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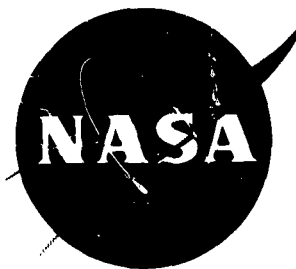
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TECHNICAL NOTE

D-989

SUMMARY OF FLIGHT-TEST RESULTS OF THE VZ-2 TILT-WING AIRCRAFT

By Robert J. Pegg

Langley Research Center
Langley Air Force Base, Va.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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SUMMARY

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Flight-test information gained from a tilt-wing research aircraft tested at the Langley Research Center has shown that design problems exist in such fields as low-speed stability and control, handling qualities, and flow separation during transition. The control power in the near-hovering configuration was considered by the pilots to be inadequate in yaw, marginal in pitch, and excessive in roll.

Solutions for some of the design problems are indicated; for example, the addition of a leading-edge droop to the wing in an attempt to delay flow separation resulted in such significantly improved handling qualities in the transition range that an additional descent capability of 1,100 feet per minute was obtained.

INTRODUCTION

The research program for vertical-take-off-and-landing (VTOL) aircraft being carried on by the National Aeronautics and Space Administration includes phases of simulator, wind-tunnel, and flight testing. Previous work attempted in some of these areas and pertaining to the VZ-2 aircraft includes: (a) studies of control response characteristics made with a variable-stability helicopter (ref. 1) which provide the basis for the control-power criteria of reference 2; (b) pilot opinion correlated to problems encountered in flying the various VTOL test beds (ref. 3); (c) time-history information of the aircraft making complete transitions and specific problem areas studied from an operational standpoint (refs. 4 to 6); (d) 1/4-scale-model free-flight and force-test data (refs. 7 to 10).

Examination of existing flight data showed a lack of satisfactory full-scale flight information on a tilt-wing VTOL-type aircraft. The tests reported herein were therefore carried out to gather documented data and pilot opinion on ground-effect characteristics, dynamic and static stability, maneuver stability, control response, and rate-of-descent limitations of the test-bed aircraft.

The investigation presented herein is aimed at bridging the gap between pilot experience and the wind-tunnel or theoretical results by presenting flight measurements of aerodynamic characteristics of the VZ-2 aircraft and interpreting the results of these measurements in terms of means for improvement for future designs.

SYMBOLS

b	wing span, ft	L
c	chord of rotor blade, in.	1
g	acceleration due to gravity, 32.2 ft/sec ²	5
I _x	moment of inertia about roll axis, slug-ft ²	7
I _y	moment of inertia about pitch axis, slug-ft ²	4
I _z	moment of inertia about yaw axis, slug-ft ²	
i _w	wing angle (measured from a line parallel to the upper longeron), deg	
M	pitching moment, ft-lb	
$\frac{\Delta M}{\Delta V}$	speed stability, $\frac{\text{ft-lb}}{\text{knot}}$	
n	number of rotor blades	
P	power, hp	
p	angular rolling velocity, radians/sec	
R	rotor-blade radius, in.	
V	airspeed, knots	
W	weight of aircraft, lb	
α_f	fuselage angle of attack, deg	

TEST EQUIPMENT

A drawing of the test aircraft including modifications is shown in figure 1, and its principal dimensions and physical characteristics are listed in table I. Figure 2 shows the test aircraft in transition flight.

Control of the aircraft in forward flight is obtained longitudinally from an all-movable horizontal-tail surface, with lateral and directional control obtained from the ailerons and rudder, respectively. In hovering flight, pitch and yaw control are obtained by varying the thrust of small fans which are located at the aft end of the fuselage. Differential collective pitch on the main rotor blades produces roll control. A phasing mechanism within the aircraft automatically varies the amount of differential collective pitch, aileron, and pitch-fan blade pitch as the wing angle is changed. The rudder and horizontal tail, however, were not connected to this phasing mechanism during this investigation and their deflections did not vary with wing angle.

Power is supplied by an 850-horsepower gas-turbine engine and is controlled by the pilot through the collective-pitch lever. Maximum usable power has been limited by various dynamic components to 650 horsepower. The aircraft has automatic stability augmentation equipment that provides rate damping about the pitch and roll axes. This equipment may be turned on or off by the pilot.

Instrumentation was provided to measure and record airspeed, altitude, rotor speed, control and wing positions, longitudinal and normal accelerations, angle of sideslip and angle of attack, engine torque, and angular velocities about the body inertia axes. Vibratory-loads measurements were also recorded during these flight tests and are presented in reference 11.

A camera mounted on the vertical tail photographed wool tufts on the right wing for visual indication of flow disturbances. A camera was also mounted on the cockpit bulkhead to photograph the control-panel instrumentation for visual monitoring of the pilot's instrument display.

DISCUSSION OF DATA

Dynamic-Stability Characteristics

Longitudinal dynamic stability.- A measure of the dynamic-stability characteristics of the aircraft is indicated by the resulting motion of the aircraft when it is disturbed by a gust or a sudden angle-of-attack

change brought on by the pilot. Figure 3 shows these characteristics by time-history plots of longitudinal pulse inputs (pitch dampers off) with the accompanying angular pitching velocities in the hovering configuration in ground effect and three forward flight conditions of 40, 62, and 100 knots. A longitudinal disturbance (pulse input) under near-hovering conditions (5-knot wind) initiates an oscillation which expands at such a rapid rate as to appear as a divergence on the first swing back through trim. Figure 3(a) shows this condition, where after approximately 3.5 seconds pitching velocity has increased to such a magnitude, with no indication of peaking, that the pilot was compelled to apply corrective action. The pilot indicated that he felt that the aircraft had zero or negative angular-velocity damping with a tendency toward simple divergence. Also, the time histories of typical pulse inputs in the forward-flight conditions, as shown in figures 3(b) to 3(d), indicated that as the wing angle is decreased the period of the oscillation is decreased and the damping of the oscillation is increased. Figure 4 shows the variation of period of longitudinal oscillation with airspeed.

Lateral dynamic stability.- In the hovering configuration ($i_w = 80^\circ$), the roll response to a pulse input is an oscillation which expands at such a rapid rate as to appear as a divergence on the first swing through the trim position (fig. 5(a)). At cruise speeds ($i_w = 9^\circ$), the oscillation is well damped, but of short period (fig. 5(b)). At the intermediate airspeeds, time-history traces were not obtained because of the directional instability of the aircraft.

Unstable Dutch roll oscillations were noted by pilots following recovery at high power from a descent condition with a wing angle of 20° . These oscillations were believed to be due to the downward inclination of the principal inertia axes with respect to the flight path following the recovery from the descent.

Static-Stability Characteristics

Static lateral-directional stability.- Two flight conditions ($i_w = 40^\circ$, $V = 40$ knots; $i_w = 9^\circ$, $V = 100$ knots) were explored to determine the static directional stability characteristics of the test aircraft. Figure 6, which is a plot of pedal position as a function of sideslip angle, shows that an instability existed in both flight regimes. This instability is indicated by a reversal in the slope of the curve between angles of sideslip of 10° and -15° for $i_w = 40^\circ$ and between angles of sideslip of -3° to 2° for $i_w = 9^\circ$. Pilot comments for other intermediate flight ranges indicate that this condition exists throughout the forward-flight speed range. At the lower airspeeds, pilots say that this instability is objectionable as a result of the deterioration of handling qualities because of increased effort and

diverted attention; however, at the higher airspeeds this condition is more objectionable because of high angular accelerations away from the desired trim and the possibility of high structural loads.

Test points taken from these same two flight conditions indicate that the variation of lateral stick position with sideslip angle showed a positive dihedral effect which pilots felt was adequate. These data are plotted in figure 7 and show that as airspeed is increased, increased dihedral effect is obtained.

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Speed stability.- The speed-stability variation of longitudinal stick position with airspeed for each of several fixed wing angles and constant power positions is shown in figure 8. The curves for airspeeds below 40 knots have been replotted from reference 4. The steepness of the slopes at the low-speed wing settings indicates that large pitching-moment changes will be experienced with inadvertent changes in airspeed, for example, in gusty air and during longitudinal oscillations. Pilot's comments indicated that flatter slopes would result in more favorable flight characteristics.

Converting the longitudinal stick position from figure 8 into pitching moment resulting from the combined tail fan and elevator control (fig. 9) shows that speed stability $\frac{\Delta M}{\Delta V}$ decreases with increase in forward speed.

Maneuver Stability

At cruise speeds where significant changes in lift can be made by changing the angle of attack, measurements of the buildup of the normal acceleration are made in a wind-up turn, which is a constant-altitude coordinated turn. At all speeds where appreciable normal acceleration results from angle-of-attack change, the early normal-acceleration response to a step input has been found to be of importance to the pilot. At the lowest speeds, when a change in the attitude of the aircraft does not produce appreciable lift change or change in flight path, only the step inputs and not the wind-up turn maneuvers are made. In a step-input maneuver, the buildup and peaking of the angular pitching velocity has been found to be a primary parameter in the study of maneuver stability.

Reference 12 outlines acceptable angular-velocity and normal-acceleration characteristics for the low-speed range for helicopters and may be considered applicable for VTOL aircraft in general.

Wind-up turn.- Figure 10 presents results from a wind-up turn at a constant velocity of 100 knots and wing angle of 9° and shows that increasing rearward longitudinal stick is required with increasing normal acceleration, as would be expected in normal airplane flight.

Step input.- Figure 11 gives time-history plots of angular pitching velocity, normal acceleration, and longitudinal stick position during step-input maneuvers. At all test conditions, the angular pitching velocity is concave downward after 2 seconds as required for helicopters in reference 12 and is considered satisfactory. The configuration with $i_w = 20^\circ$ (fig. 11(a)) is the only configuration of those shown which shows any discernible change in normal acceleration (initial positive buildup limited by stall). At this particular flight configuration, abrupt flow breakdown occurs over the wing causing a nose-down pitching moment and a reduction in acceleration and angular pitching velocities at the onset of stall.

Variation of Power Required With Airspeed

The variation of wing angle of attack with airspeed is shown in figure 12 and includes data where the fuselage angle of attack was varied up to 10° . Figure 13 shows the power required for various level-flight airspeeds of the aircraft before and after wing modification. Data were obtained at $\alpha_f \approx 0^\circ$ both before and after the aircraft wing was modified, with no critical change except a slight decrease in the power required as a result of unstalling the wing. Power measurements presented at wing angles greater than 60° were obtained near the ground because of the lack of rotor horsepower available to hover out of ground effect. All power readings indicate the total engine output and not actual rotor horsepower.

Control

The control power, which is of primary importance in the handling qualities of an aircraft, is defined herein as the moment on the aircraft produced for a given control displacement. The control power in the near-hovering configuration was considered by the pilots to be inadequate in yaw, marginal in pitch, and excessive in roll.

Roll control power.- Roll control power in the hovering configuration is about twice that of the cruise configuration as shown in figure 14. Pilots have objected to the excessive roll control power in the hovering configuration obtained with the present linkage arrangement. Internal mechanical changes can be made to the linkages to vary roll control power, but have not been made during this series of tests. From the hovering position to full wing-down position, roll control

per inch of stick goes from $1.08 \frac{\text{radians/sec}^2}{\text{in.}}$ to approximately $0.6 \frac{\text{radians/sec}^2}{\text{in.}}$.

Maximum roll velocities induced by the pilot in hovering flight on the test aircraft are greater than desirable. Figure 15 shows the rolling velocity per inch of lateral stick deflection plotted against airspeed. Objectionable rates of $32 \frac{\text{deg/sec}}{\text{in.}}$ are noted in hovering while acceptable and desirable values of $10.9 \frac{\text{deg/sec}}{\text{in.}}$ are obtained at 100 knots. This value corresponds to $\text{pb}/2V = 0.068$ at the high-speed condition with full lateral stick deflection. These records were taken with roll dampers turned off.

Yaw control.- The variation of yaw-fan static thrust with pedal displacement is shown in figure 16. This nonlinear control characteristic, particularly near zero fan thrust, is objectionable to pilots. The average maximum yaw control power provided by the fan, experimentally determined in hovering, is about 885 foot-pounds. This moment was insufficient to handle the random yawing motions produced by the flow characteristics near the ground. For a test vehicle of this type, little hazard results from the lack of yaw control in hovering; however, for operational vehicles intended to perform precision maneuvers, the control-power requirements of reference 12 are firmly recommended. This subject is discussed more completely in reference 2.

Trim

Trim change with airspeed.- The variation of longitudinal stick position with trim airspeed is shown in figure 17. Two conditions were explored: one at a fuselage angle of attack of 0° and the other at varying fuselage attitudes. In trim unaccelerated flight, 30 percent of the total longitudinal stick travel was used to maintain a constant fuselage angle of attack of 0° . Pilots commented, however, that large trim changes were not noticed at airspeeds below 35 knots. In normal practice, the aircraft was usually flown through the speed range with the fuselage attitude varying somewhat. The only trim change which was objectionable under these conditions occurred during a rapid conversion from a wing angle of approximately 20° to 9° and back. This change is shown in figure 8.

Trim change with power.- A definite trim change with power is experienced by the aircraft during transition. Test points taken at two wing angles showing the variation of longitudinal stick position for trim with power are given in figure 18. The curve representing trim change with power at a wing angle of 40° is an example of the steepest gradient encountered by this aircraft, but pilots indicate that it is within tolerable limits although it is more desirable to have no trim change. At a wing angle of 25° , the trim longitudinal stick position

changes approximately 15 percent of the total travel from a rate of climb of 1,400 feet per minute to a rate of descent of 1,300 feet per minute.

Ground Effect

Figure 19 shows the behavior for the tilt-wing configuration in a near-hovering condition in and out of ground effect without automatic stabilization. Figure 19(a) illustrates moderate aircraft and control motions out of ground effect. For the aircraft in ground effect (fig. 19(b)), aircraft and control motions are many times greater, with erratic angular-velocity changes up to 10 degrees per second and frequent control motions of several inches. The effects of the ground have been noted with the aircraft wheels as high as 19 feet above the ground. The problem of erratic aircraft motion in ground effect may be expected to arise in practice for a variety of designs, especially when the aircraft is operating over rough ground, in gusty air, or when it is not maintaining a level attitude of the wings. Buffeting is apt to be encountered, also, in a variety of designs.

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Conversion

Data have been obtained for full conversions from hovering to forward flight and back to hovering, and a typical resulting time-history plot is presented in figure 20. In general, take-off, landing, and the low-speed portion of the conversions have caused the pilots most concern; however, the conversion maneuver presents no great difficulty to the pilots.

Rate-of-Descent Limitations

The most critical region of operation for the VZ-2 is encountered, as a result of wing stall and separation, in decelerating conversion and/or descent. This wing stall and separation leads to buffeting and erratic motions, with general difficulty in handling the aircraft. This stall results in regions of the expected VTOL velocity-rate-of-climb envelope being completely unacceptable for normal flight operations. The limits of the flight envelope have been defined by pilot opinion and are presented in figure 21.

In an attempt to alleviate the flow separation, a modification to the leading edge was made. The modification consisted of additional thickness near the leading edge and an increased leading-edge radius and resulted in approximately 6° of leading-edge droop. This modification is illustrated in figures 1(b) and 1(c).

Figure 21 shows the successive shifts in the rate-of-descent boundaries for the addition of partial-span and full-span droop. It is not to be implied from one success with this device that a thorough understanding of this flow-separation problem has been attained. The leading-edge droop, as such, was not expected to be nearly so effective, and the changed relative position of the wing and propeller axis may have had a material effect on the results.

Basic wing.- Figure 21(a) presents the rate-of-descent limitations for the basic wing condition. In level flight, as the wing is raised and the airspeed is decreased from 75 knots to approximately 60 knots ($i_w = 20^\circ$), abrupt flow breakdown occurs over the wing and causes wing dropping or aircraft roll-off accompanied by buffeting, yaw disturbances, and a nose-down trim change (requiring rearward-stick displacement to offset).

In a constant-descent condition, as the wing angle is increased from 30° to 50° and the aircraft airspeed drops from 55 knots to 35 knots, limiting stall conditions are not as severe as those found at lower wing angles because of the decreased dynamic pressure; but considerable directional and longitudinal disturbances and a reduction in control effectiveness make flying in the shaded areas equally undesirable. At wing angles greater than 50° , the limiting feature seems to arise chiefly from the general lack of control effectiveness about the directional and longitudinal axes of the aircraft.

Basic wing with droop.- The first improvement, which is shown in figure 1(b), was accomplished through the addition of a drooped leading edge over the outboard portion of the wing. This modification increased the rate-of-descent capability at an airspeed of approximately 60 knots by approximately 600 feet per minute (fig. 21(b)). The second improvement, which increased the maximum rate of controllable descent at this same airspeed by approximately another 500 feet per minute (fig. 21(c)), was due to the further addition of inboard drooped leading edges. Other modifications, such as the inboard wing fences illustrated in figure 1 and aileron droop, indicated no significant improvement over that shown by the full-span drooped leading edge.

The leading-edge droop not only lowered the "unacceptable" boundaries, but it also made flying in the "poor" areas much easier.

It should be pointed out that in a decelerating condition in level flight, the boundary for the equivalent descent rate indicates the region of unacceptable flight characteristics; that is, the boundaries of figure 21 are effectively raised by deceleration. Therefore, in level flight the conversion rate can also be limited by wing stall and separation.

CONCLUSIONS

Pertinent results of the flight-test investigation of the VZ-2 tilt-wing VTOL/STOL aircraft indicate the following conclusions:

1. Pitch and roll pulse inputs initiated an oscillation which expanded at such a rapid rate as to appear as a divergence on the first swing through the trim position.
2. The aircraft shows increasing positive speed stability with decreasing airspeed, a condition which can cause large variations in the pitching moment with inadvertent changes in airspeed.
3. Hovering control power of the aircraft is considered by the pilot to be inadequate in yaw, marginal in pitch, and excessive in roll.
4. Ground interference causes erratic aircraft motions which, without the use of automatic stabilization, limit operation when the aircraft wheels are within 19 feet of the ground.
5. Wing stall and separation leading to buffeting, erratic motions, and general difficulty in handling the aircraft, result in the desired VTOL velocity-rate-of-climb envelope having regions completely unacceptable for normal flight operations. The addition of a full-span leading-edge droop decreased the regions that were unacceptable for normal flight and thereby permitted an additional 1,100 feet per minute descent capability at an airspeed of approximately 60 knots.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Air Force Base, Va., September 25, 1961.

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TABLE I.- PHYSICAL CHARACTERISTICS OF THE VZ-2 AIRCRAFT

Rotors:

Diameter, ft	9.5
Blade chord, in.	13
Blade twist (linear, root to tip), deg	19.2
Airfoil section	NACA 0009 with 0.5-in. cusp
Blade taper ratio	1
Solidity ($\frac{nc}{\pi R}$)	0.218
Distance between propeller axes, ft	14.67
Operational speed, rpm	1,416
Differential pitch, deg	± 2

Wing:

Span (excluding tips), ft	24.88
Chord, ft	4.75
Airfoil section	NACA 4415
Taper ratio	1
Sweep, deg	0
Dihedral, deg	0
Pivot, percent chord	37.6
Ailerons -	
Chord, ft	1.25
Span, ft	5
Tilt range (referenced to upper longeron), deg	9 to 85

Vertical tail:

Height, ft	5.43
Approximate mean geometric chord, ft	5.90
Sweep at leading edge, deg	28
Basic airfoil section	NACA 0012
Rudder -	
Chord, in.	21.5
Span, in.	58.0

Horizontal tail:

Span (less tips), ft	9.90
Chord, ft	3.00
Sweep, deg	0
Taper ratio	1
Airfoil section	NACA 0012
Dihedral, deg	0
Length (distance from wing pivot to leading edge of tail), ft	10.475
Hinge point (distance from leading edge), in.	8.3

TABLE I.- PHYSICAL CHARACTERISTICS OF THE VZ-2 AIRCRAFT - Concluded

Control fans:

Diameter (both fans), ft	2.00
Moment arm about wing pivot (both fans), ft	12.35
Number of blades	4
Rotor speed, rpm	5,850
Fuselage length, ft	26.4
Engine	Lycoming T53
Weight as flown with ejection seat, lb	3,500
Center of gravity (for 9° wing incidence), percent M.A.C.	33.5
Center of gravity (for 85° wing incidence), ft forward of pivot point (measured along longitudinal axis)	0.135

Moments of inertia:

Aircraft weight = 3,432 lb -

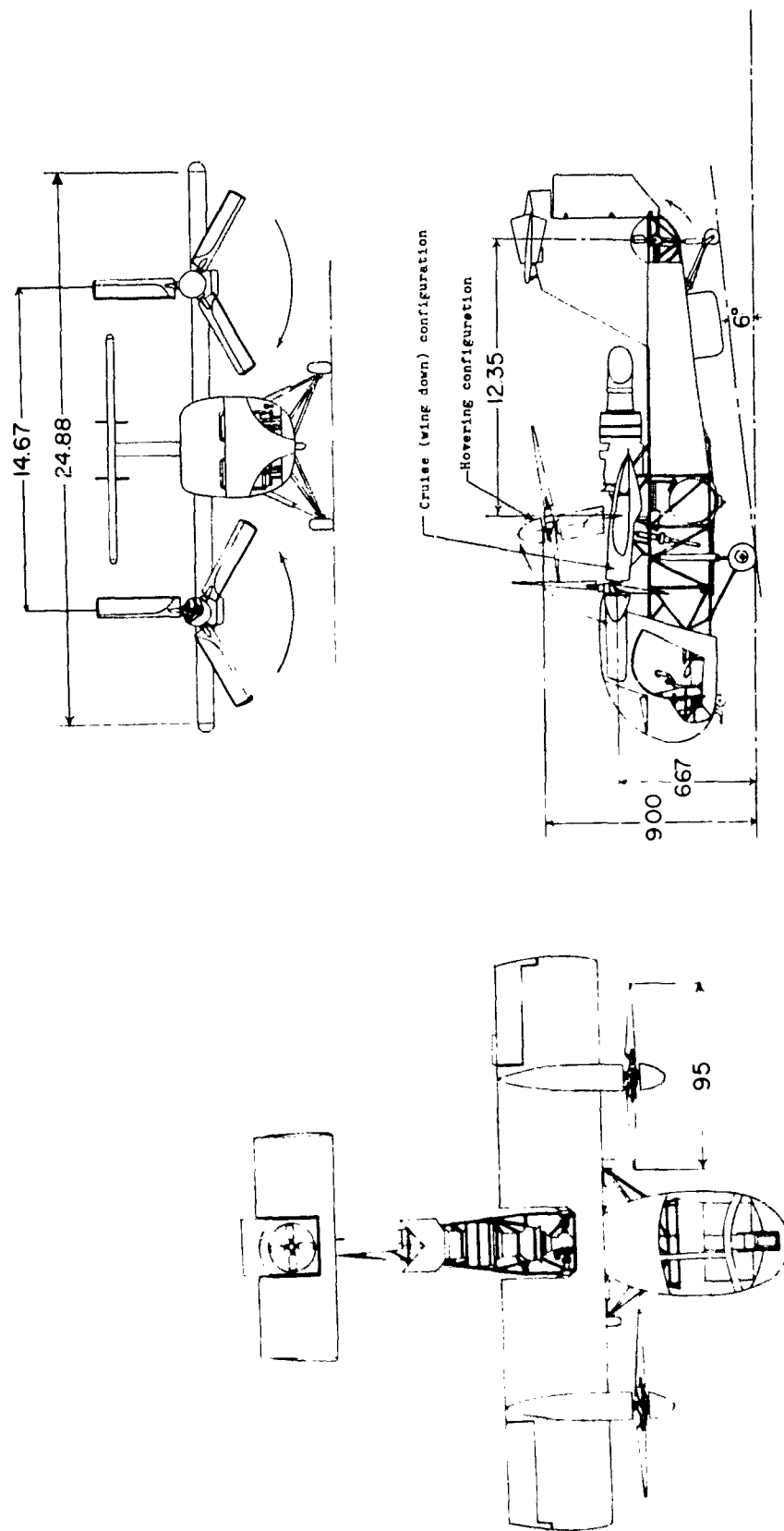
I_X , slug-ft ²	1,634
I_Y , slug-ft ²	2,937
I_Z , slug-ft ²	3,988

Aircraft weight = 3,204 lb -

I_X , slug-ft ²	1,560
I_Y , slug-ft ²	2,899
I_Z , slug-ft ²	3,985

Total control travels:

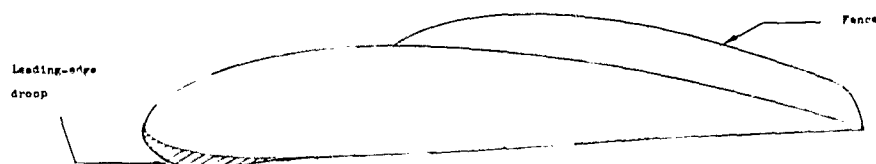
Lateral stick, in.	$9\frac{1}{8}$
Longitudinal stick, in.	$11\frac{1}{8}$
Pedal, in.	6



(a) Sketch of tilt-wing VTOL aircraft. All dimensions are in feet unless otherwise specified.

Figure 1.- Sketches of VZ-2 aircraft including wing modifications.

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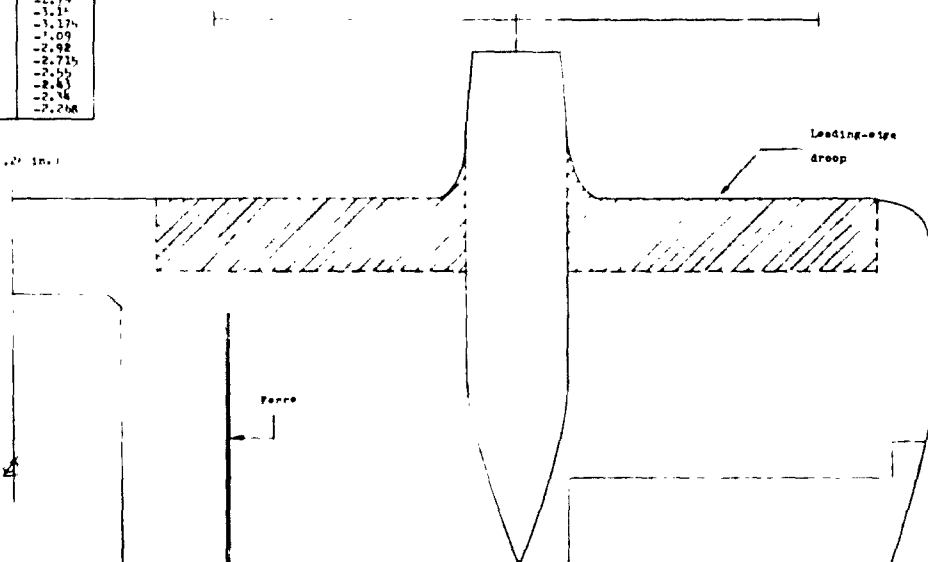


(b) Airfoil section showing wing fence and droop leading edge.

Station, in.	Wing modification coordinates, in. -	
	Upper ordinate	Lower ordinate
0.7125	1.75	-2.28
1.425	2.178	-1.79
2.137	3.272	-1.11
2.850	3.08	-1.17
3.562	4.444	-1.09
4.275	—	-2.92
4.987	5.283	-2.715
5.700	—	-2.55
6.412	4.443	-2.44
7.125	—	-2.44
7.837	6.224	-2.268

L.F. radius, 2.878 in.

Center at (2.461 in., -1.22 in.)



(c) Wing planform showing location of fences and leading-edge droops.

Figure 1.- Concluded.

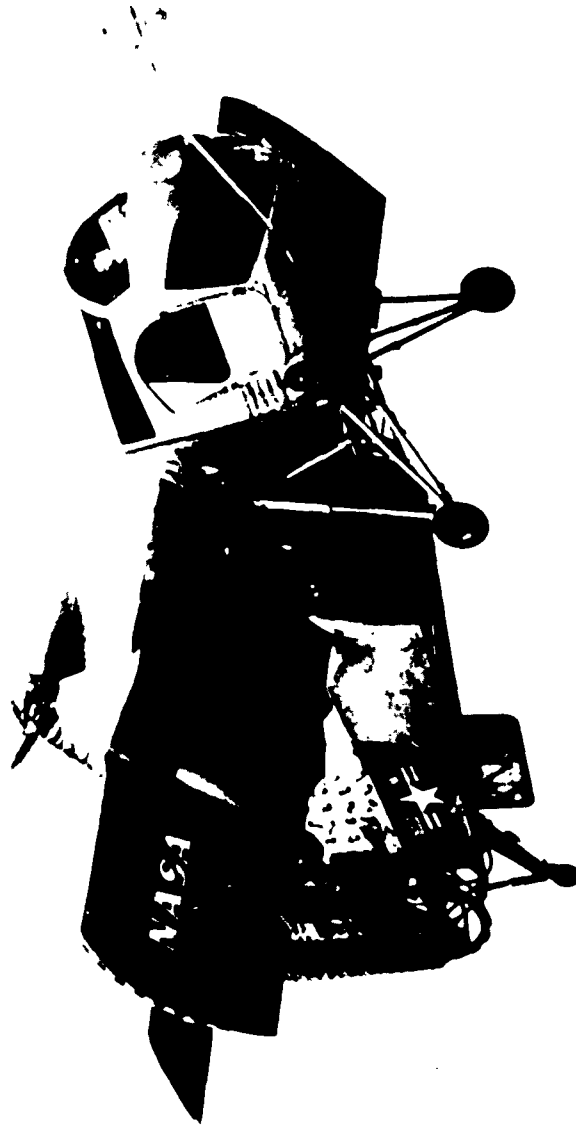
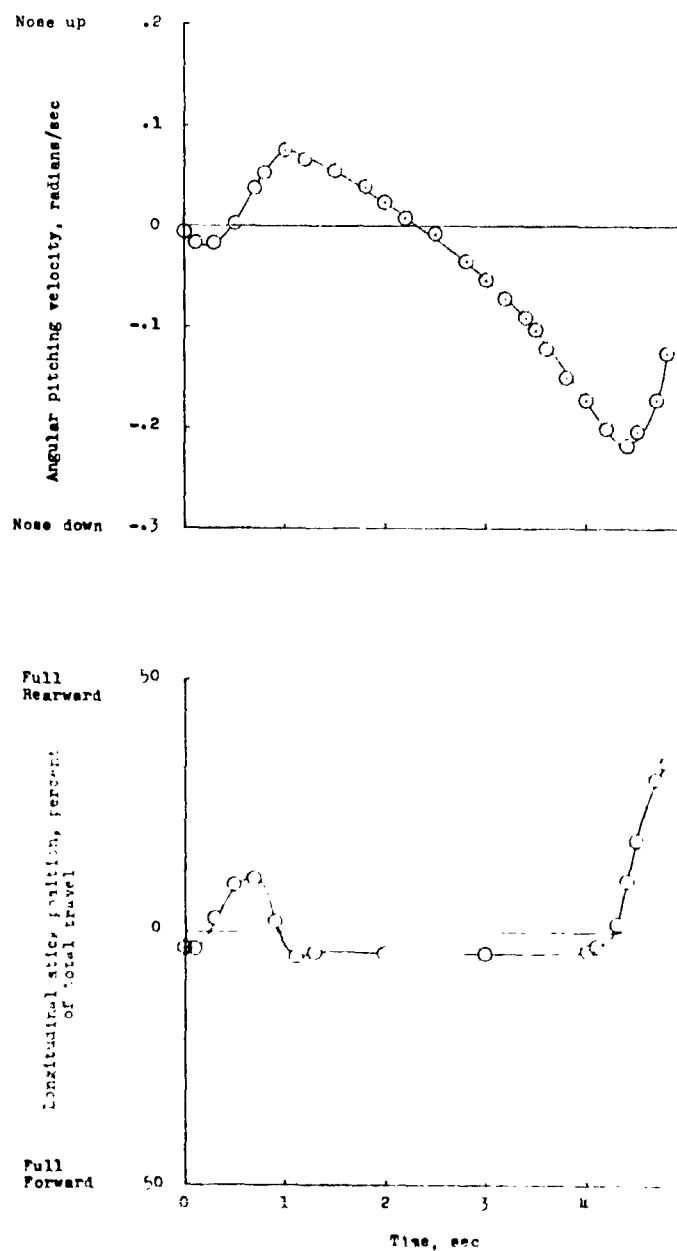


Figure 2.- Test aircraft in transition flight.

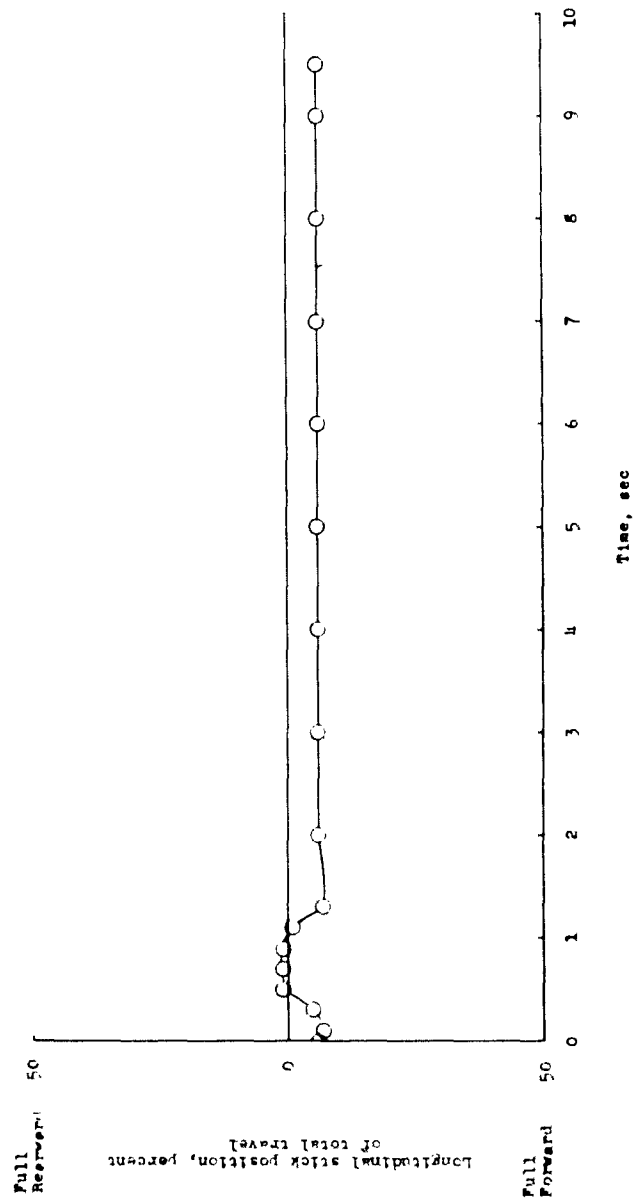
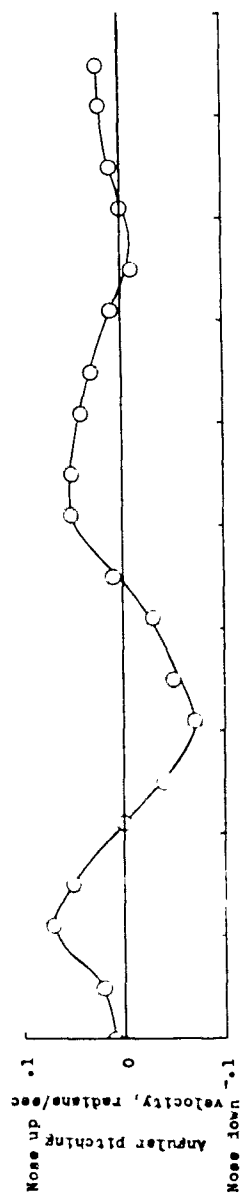
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(a) Longitudinal pulse input in the hovering configuration in ground effect. 5-knot headwind; $i_w = 80^\circ$; $P = 630$ hp.

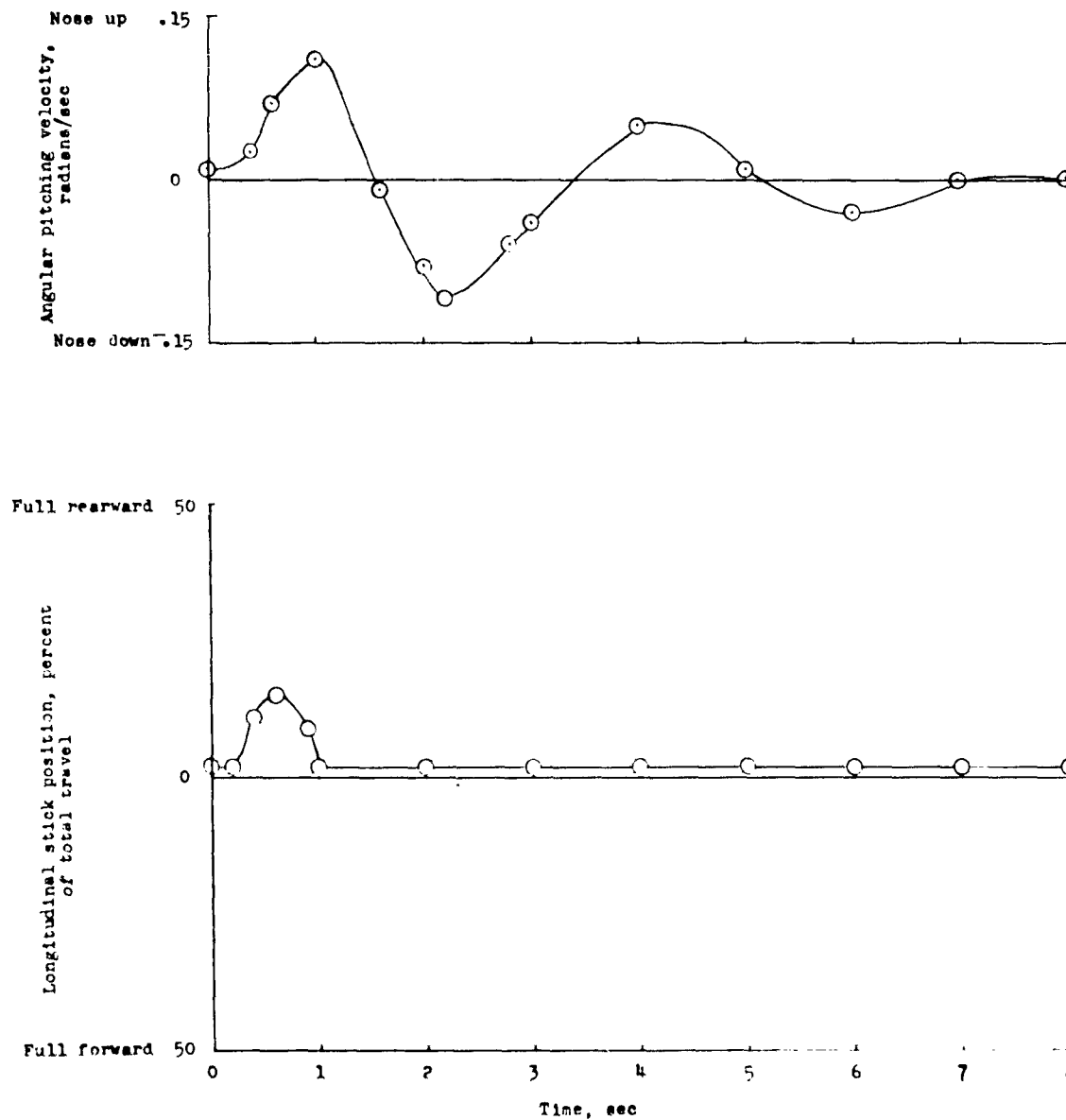
Figure 3.- Longitudinal dynamic-stability characteristics in the hovering and transition flight configurations measured with pitch dampers off.



(b) Longitudinal pulse input at forward velocity of 40 knots. $I_w = 40^\circ$; $P = 525$ hp.

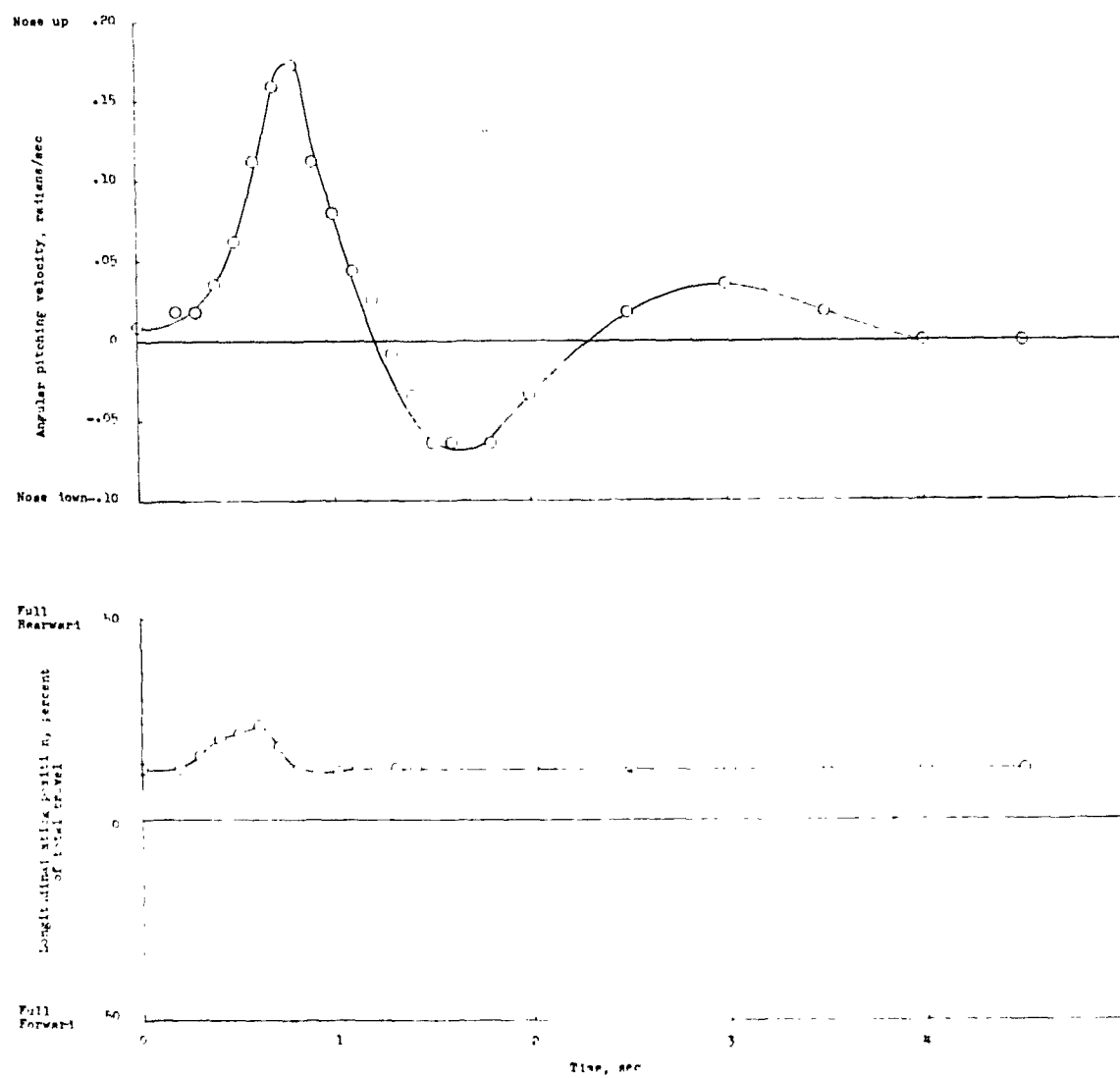
Figure 3.- Continued.

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(c) Longitudinal pulse input at forward velocity of 62 knots.
 $i_w = 20^\circ$; $P = 450$ hp.

Figure 3.- Continued.



(d) Longitudinal pulse input at forward velocity of 100 knots.
 $i_w = 9^\circ$; $P = 600$ hp.

Figure 3.- Concluded.

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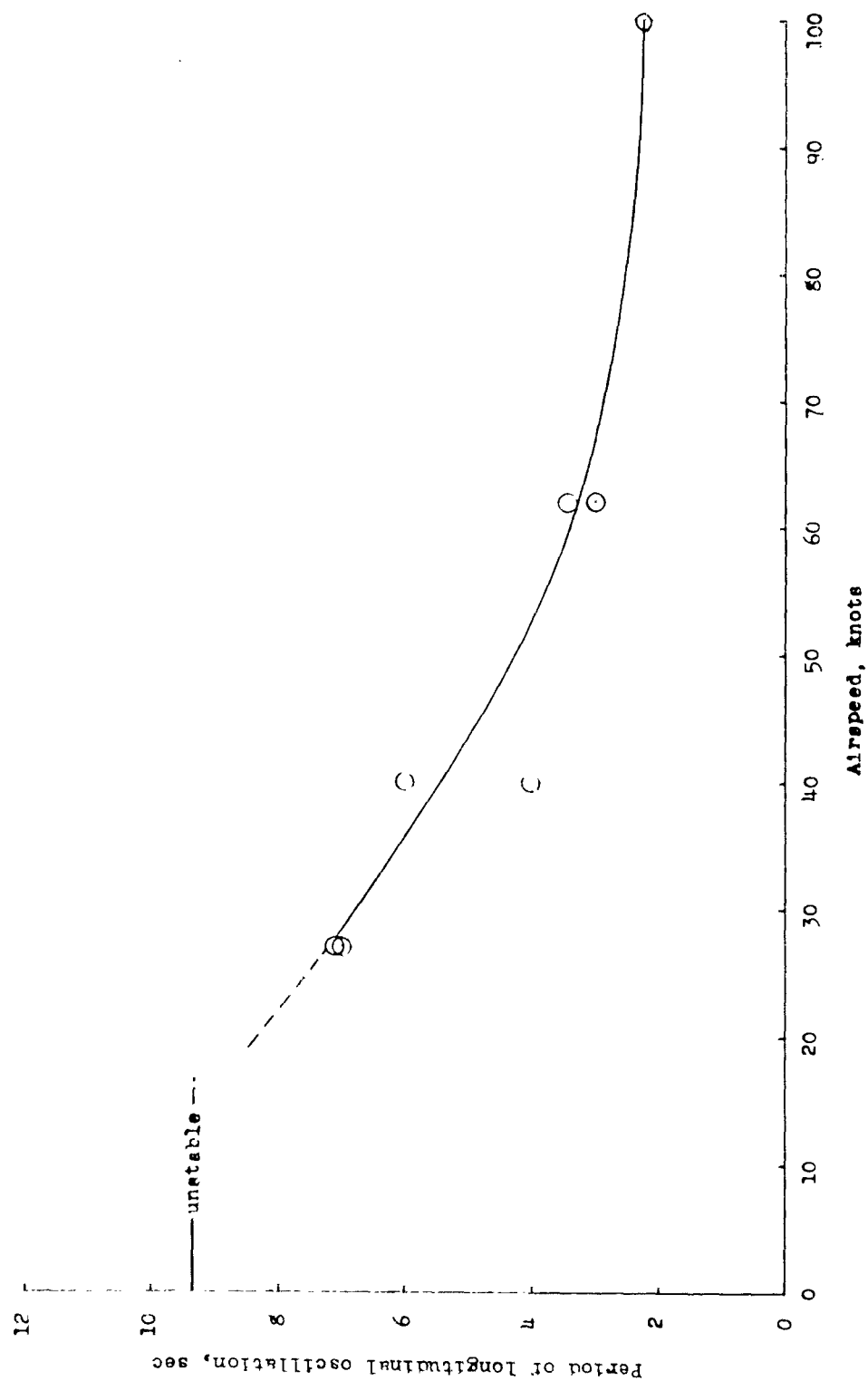
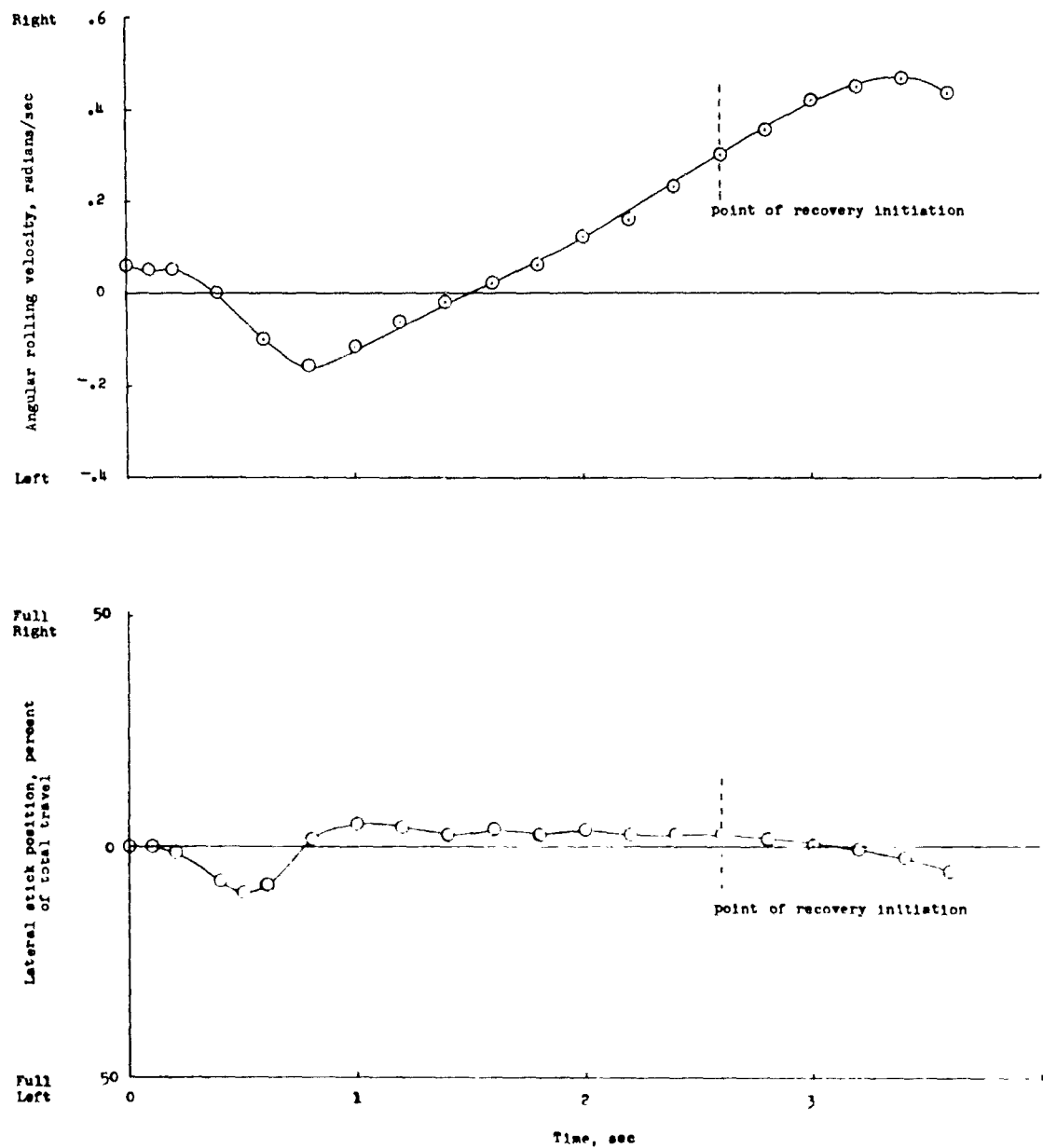


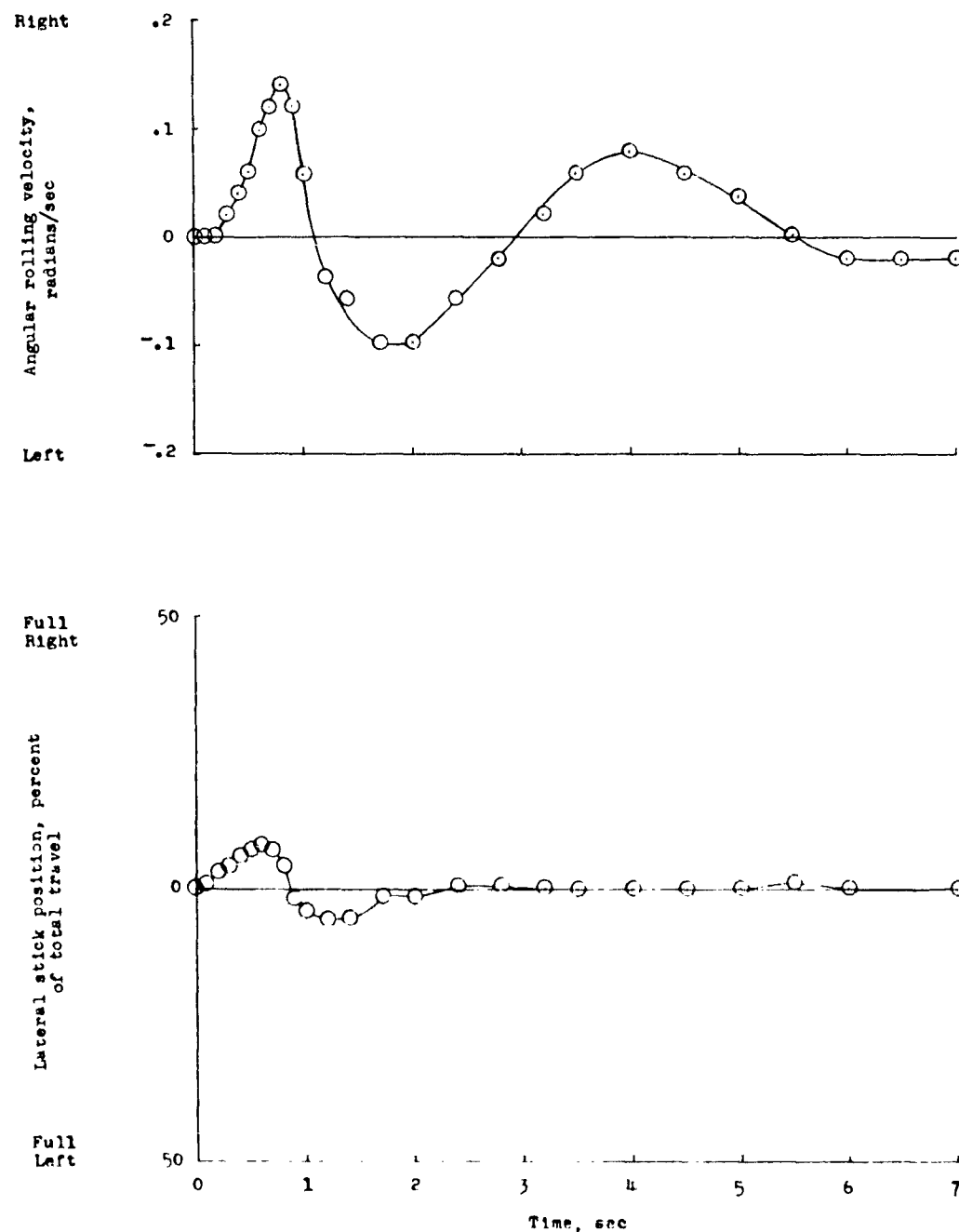
Figure 4.- Period of longitudinal oscillation due to a longitudinal pulse input while in approximately level flight plotted against airspeed. Pitch dampers off.



(a) Lateral pulse input while hovering into a 5-knot headwind.
 $i_w = 80^\circ$; $P = 630$ hp.

Figure 5.- Lateral dynamic stability characteristics in the hovering and cruise configurations measured with roll dampers off.

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(b) Lateral pulse input at forward velocity of 99 knots.
 $i_w = 90^\circ$; $P = 530$ hp.

Figure 5.- Concluded.

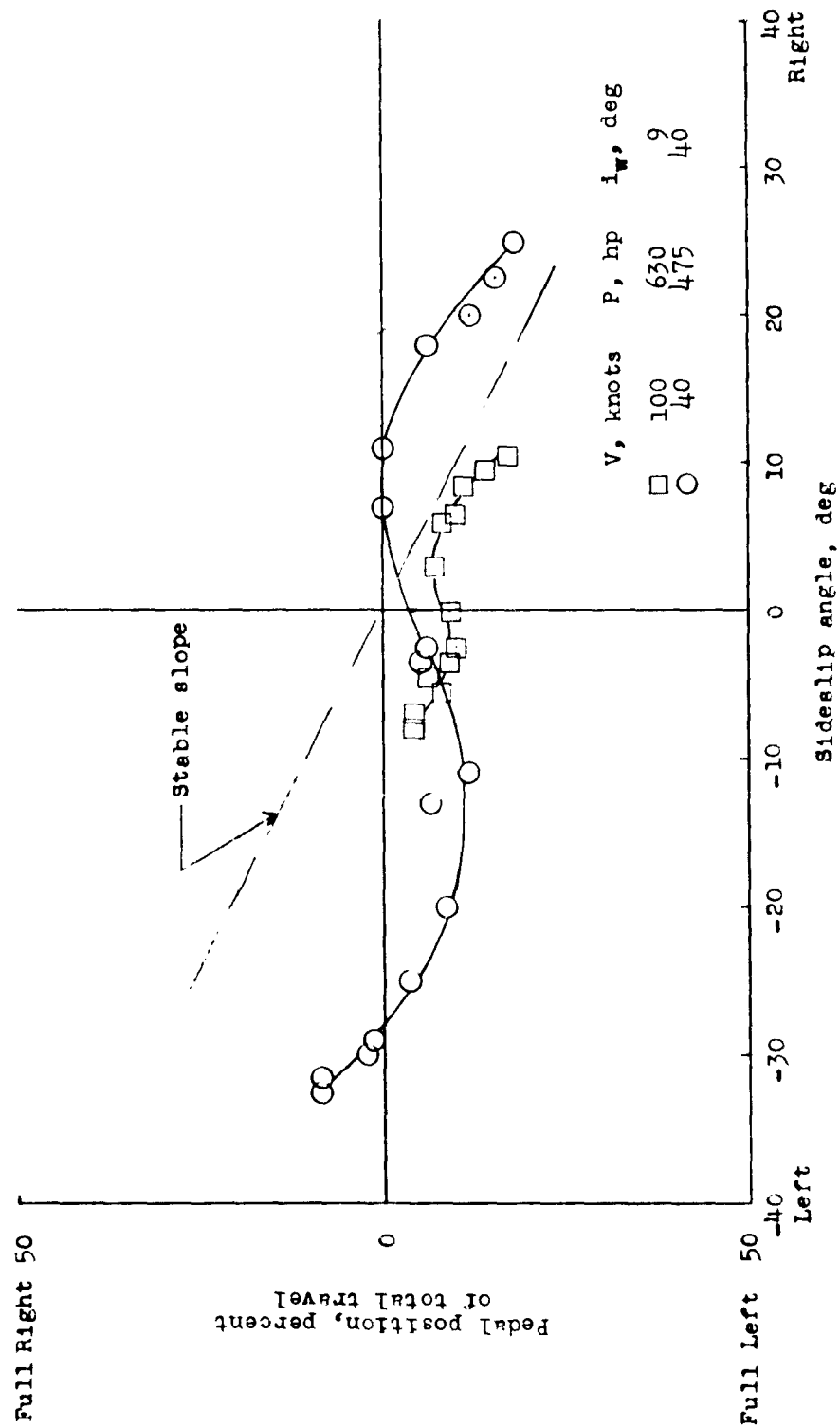


Figure 6.- Static directional stability characteristics. Pedal position as a function of sideslip angle. Dampers off.

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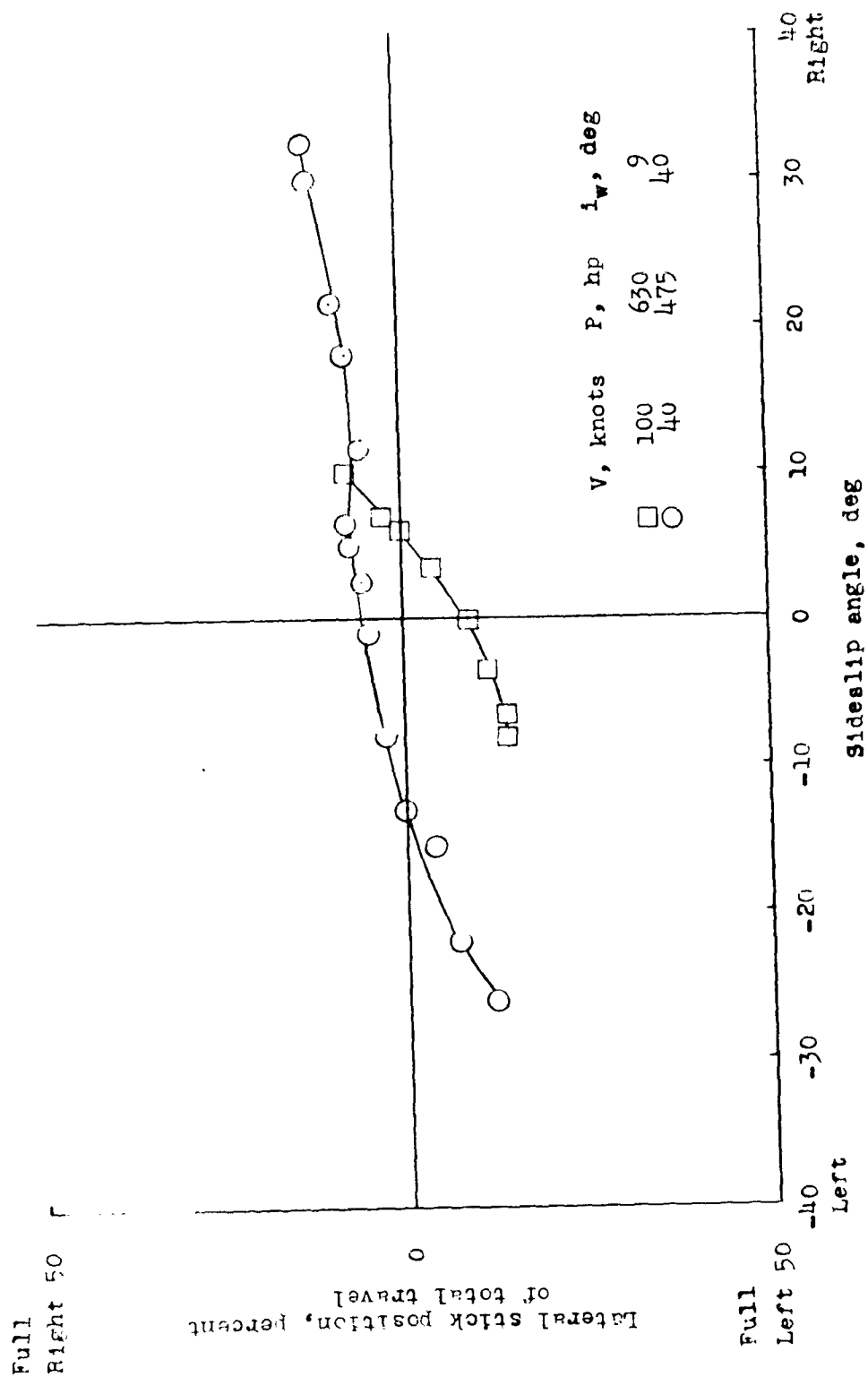


Figure 7.- Dihedral effect on the test aircraft at two forward flight conditions. Dampers off.

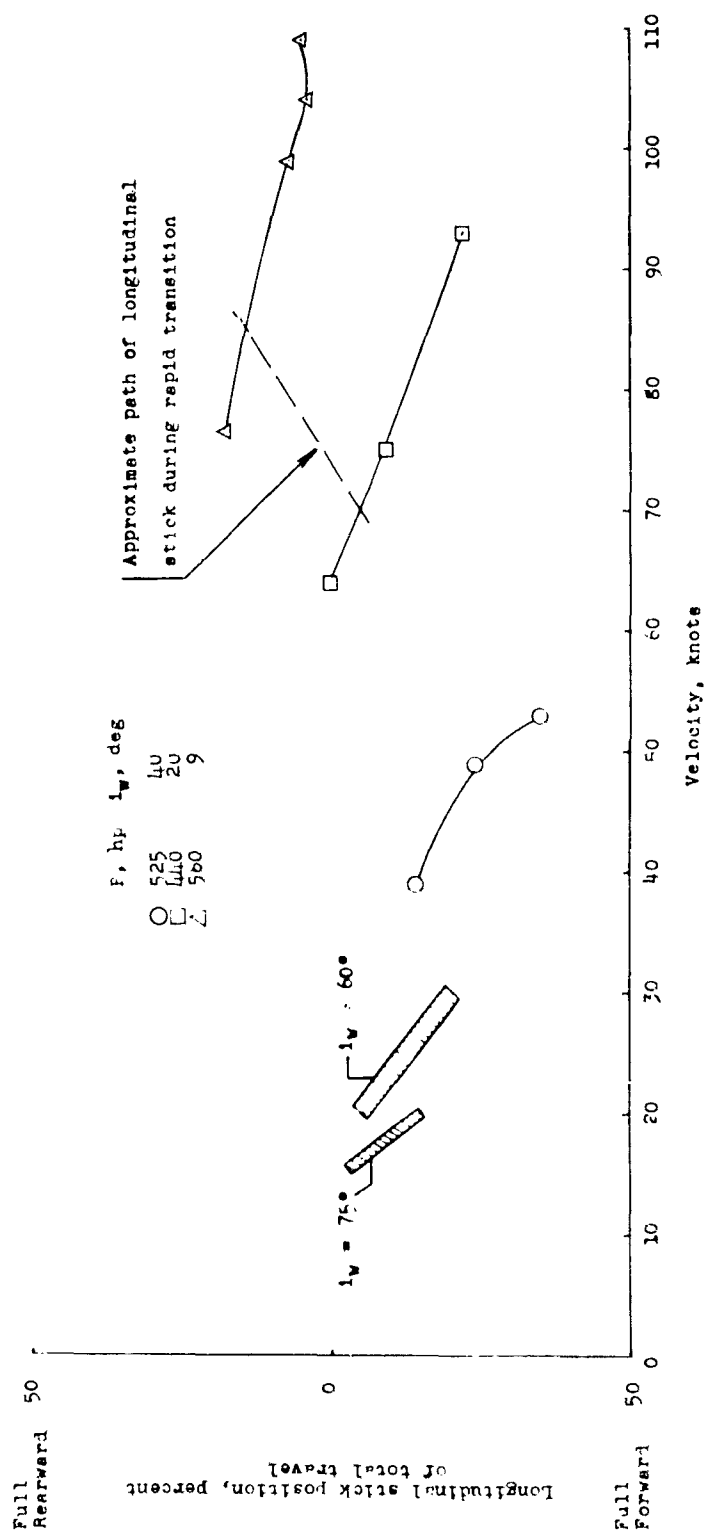


Figure 8.- Variation of trim longitudinal stick position with airspeed for constant power at various wing angles. Shaded bands are from reference 4 where no attempt was made to hold power constant.

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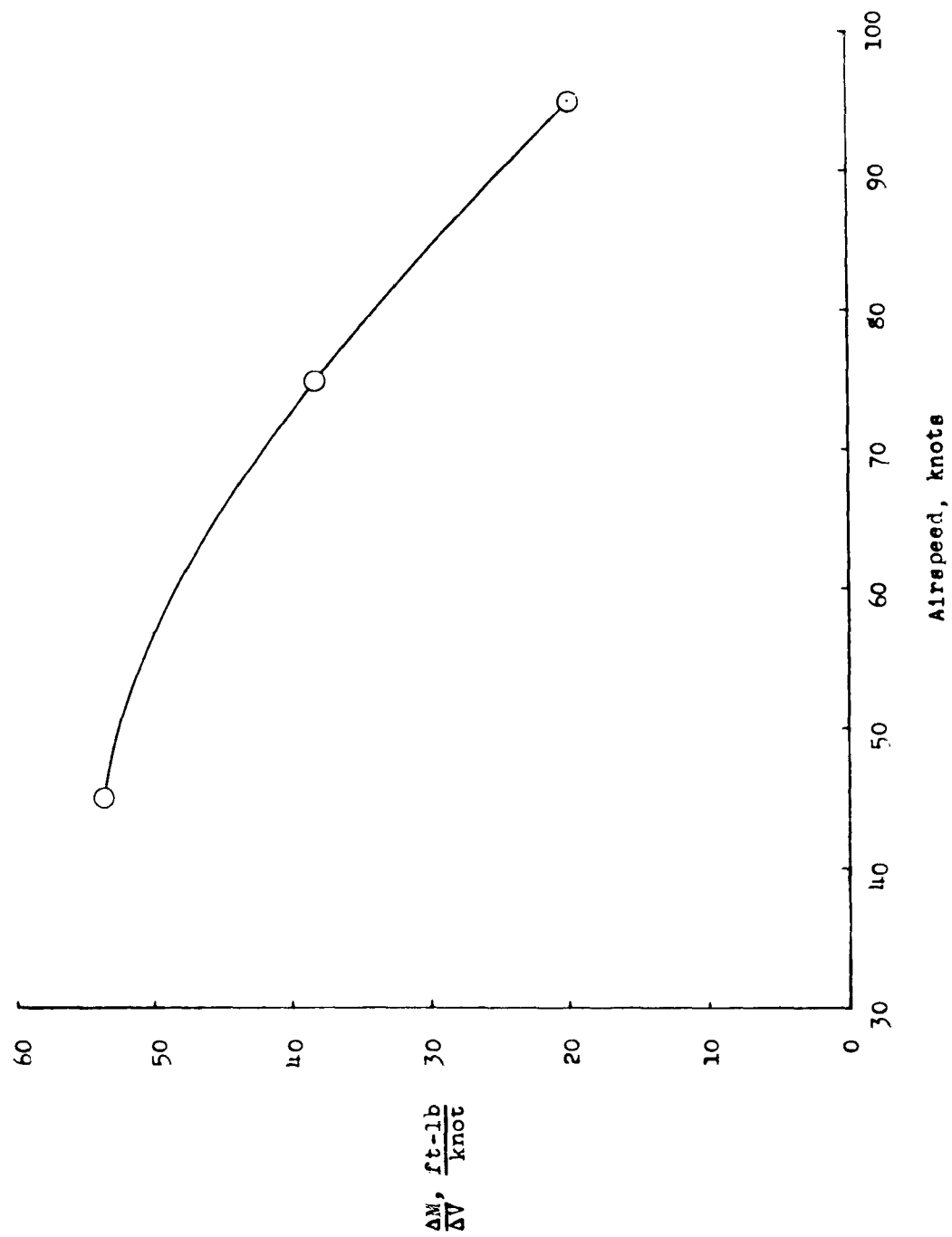


Figure 9.- Variation of speed stability with airspeed.

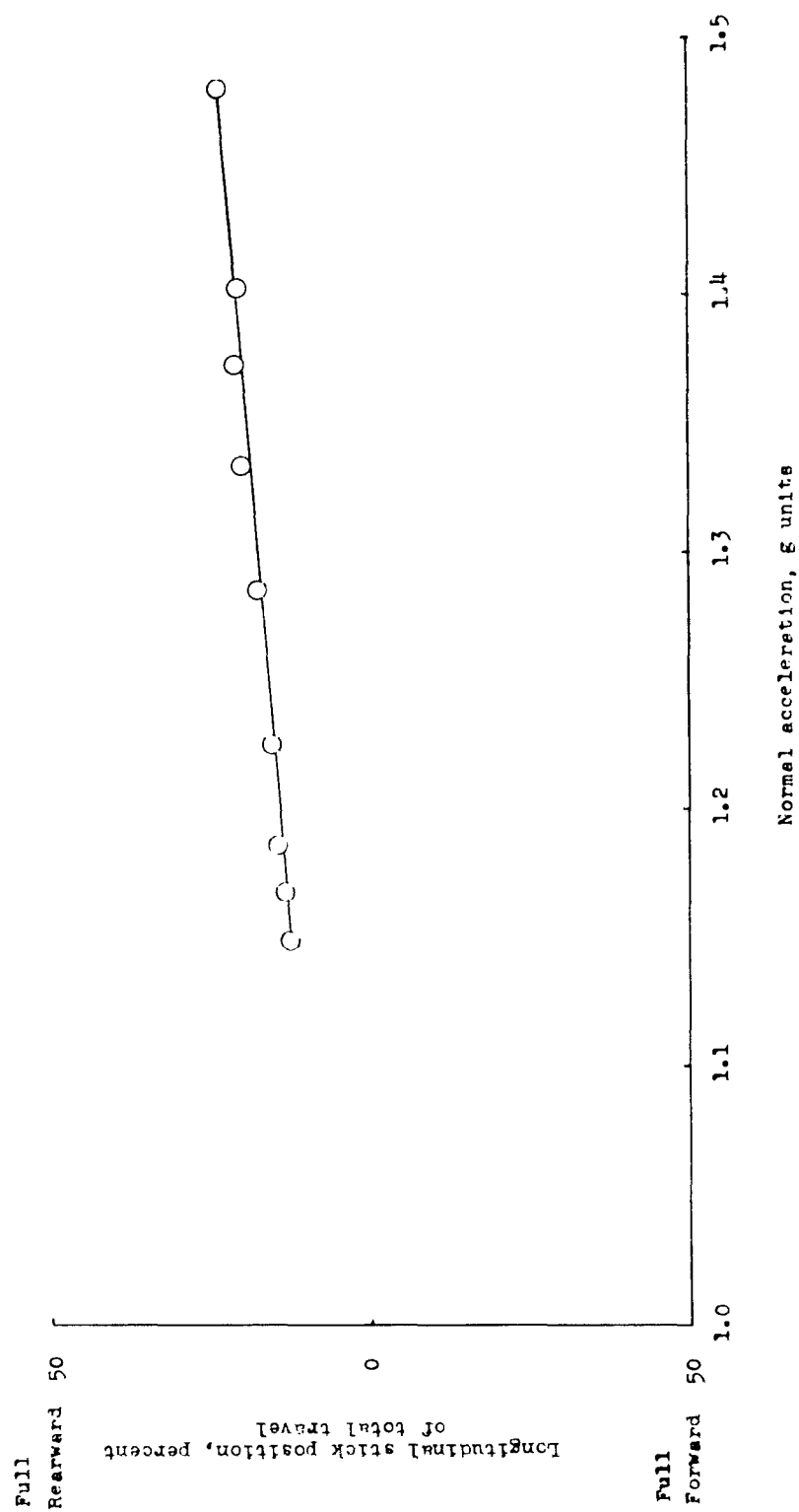
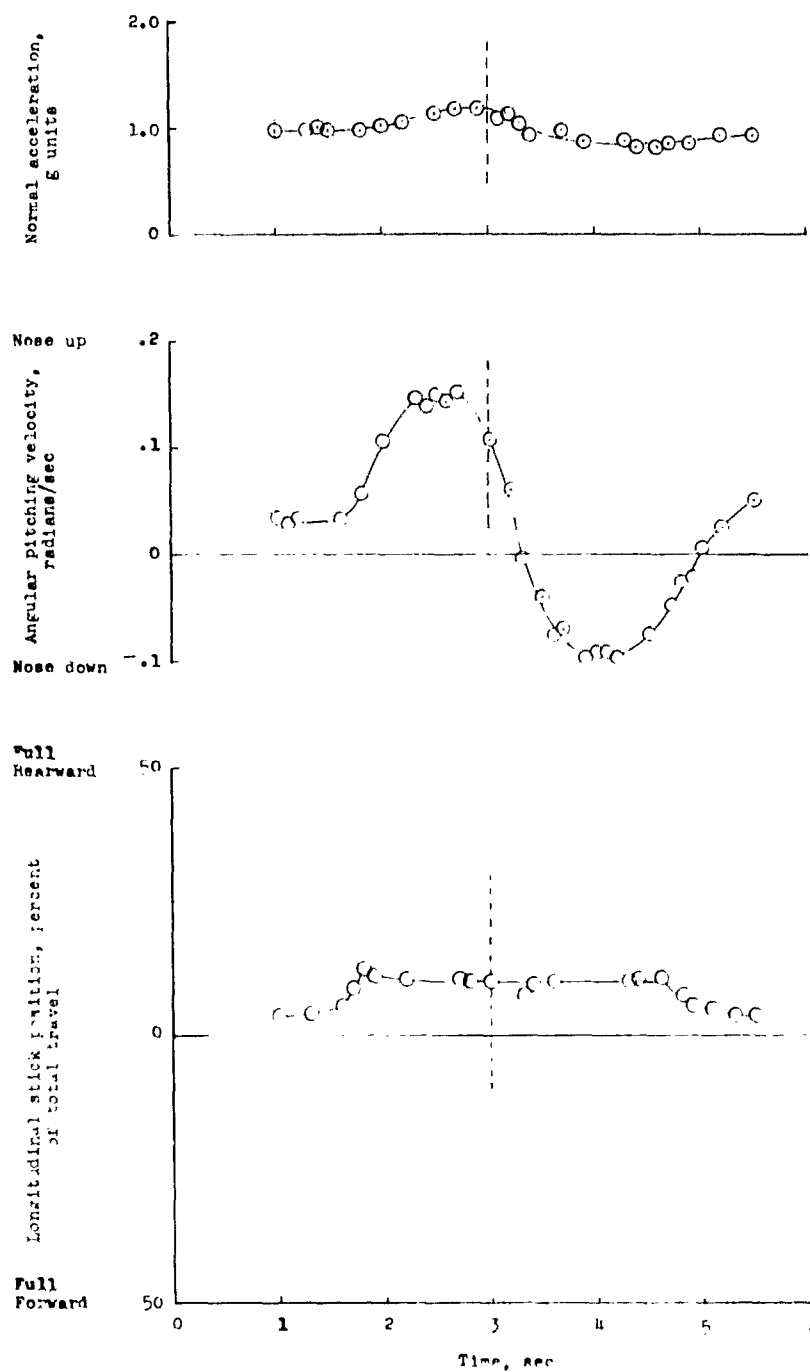


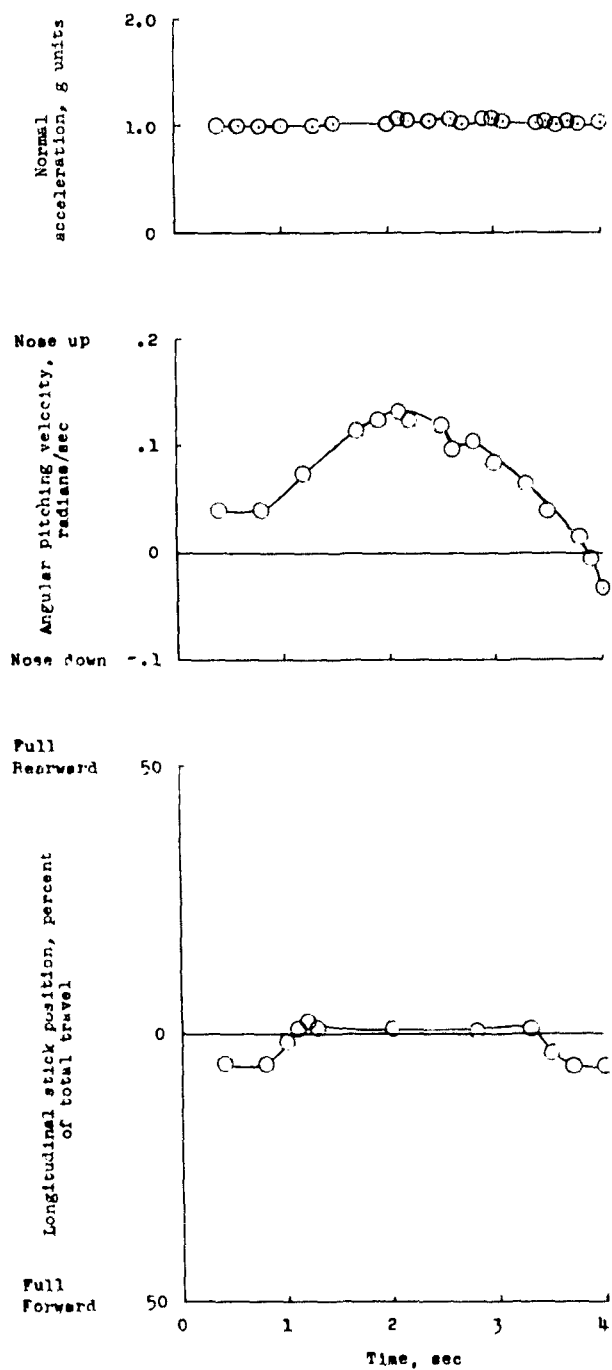
Figure 10.- Results of a wind-up turn at 100 knots with dampers off. $i_w = 9^\circ$; $P = 630$ hp.

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(a) $V = 62$ knots; $P = 450$ hp; $i_w = 20^\circ$.

Figure 11.- Time-history records of normal acceleration and angular pitching velocity resulting from a longitudinal control input at three different flight conditions.

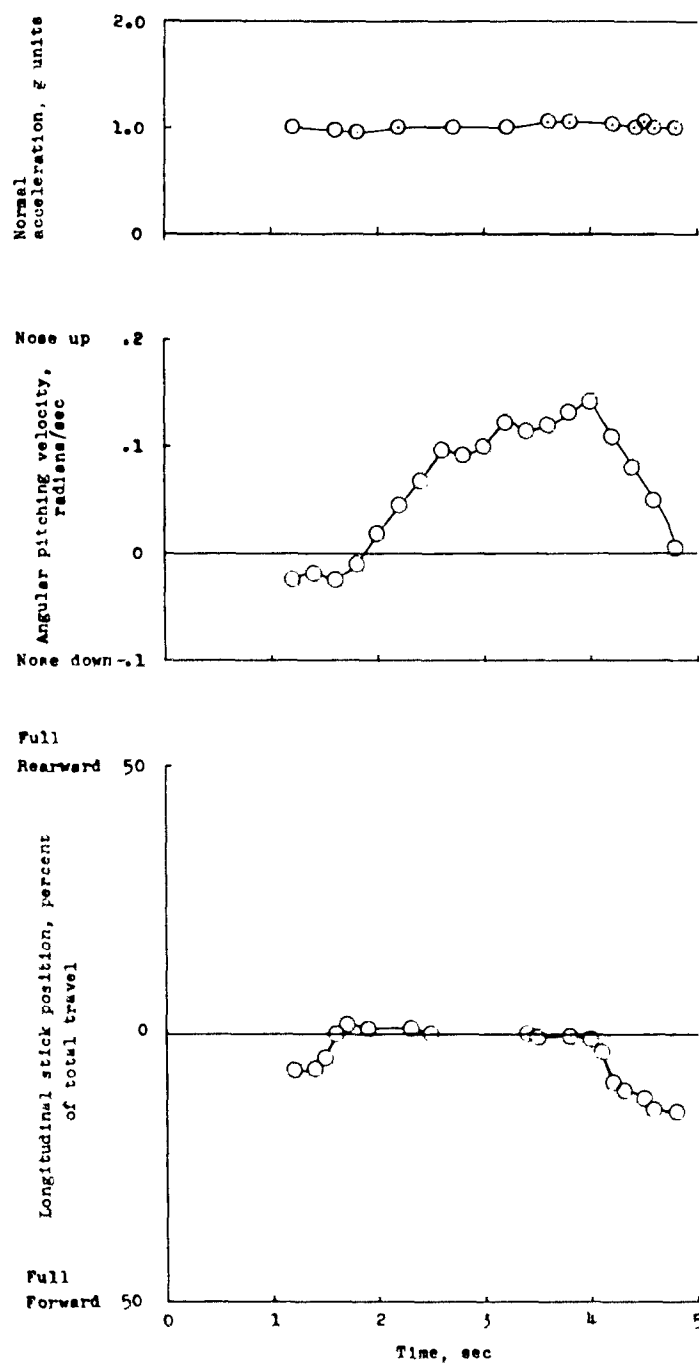


(b) $V = 40$ knots; $P = 525$ hp; $i_w = 40^\circ$.

Figure 11.- Continued.

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(c) $V = 27$ knots; $P = 550$ hp; $i_w = 50^\circ$.

Figure 11.- Concluded.

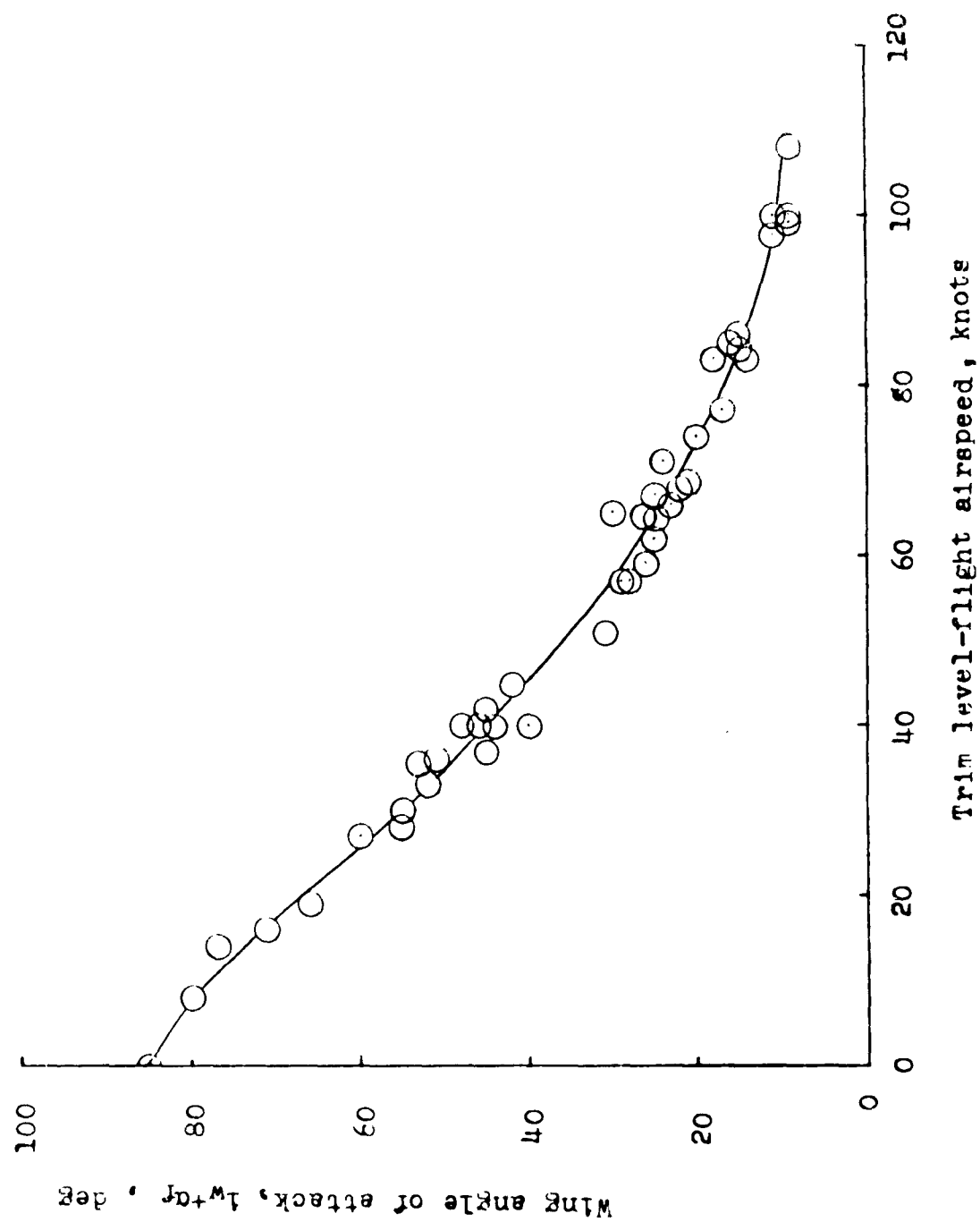


Figure 12.- Variation of wing angle of attack with airspeed.

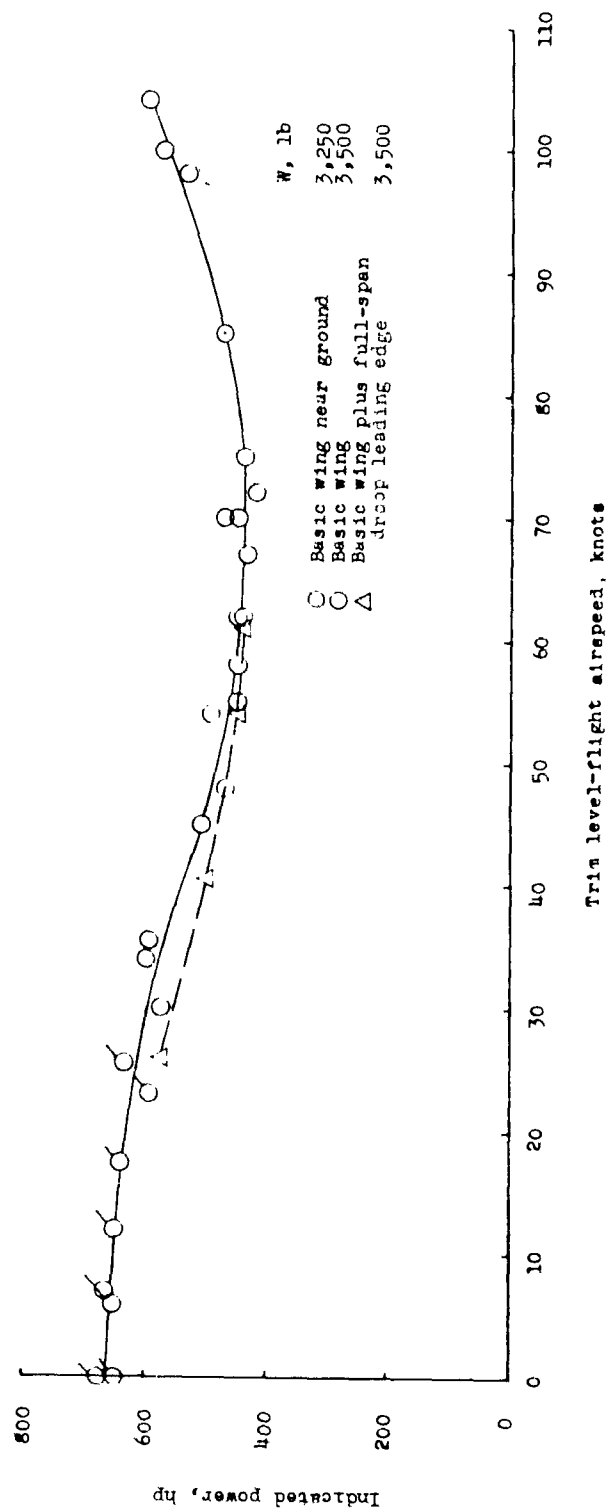


Figure 13.- Power-required curve of the test aircraft before and after wing modification.
 $\alpha_f \approx 0^\circ$.

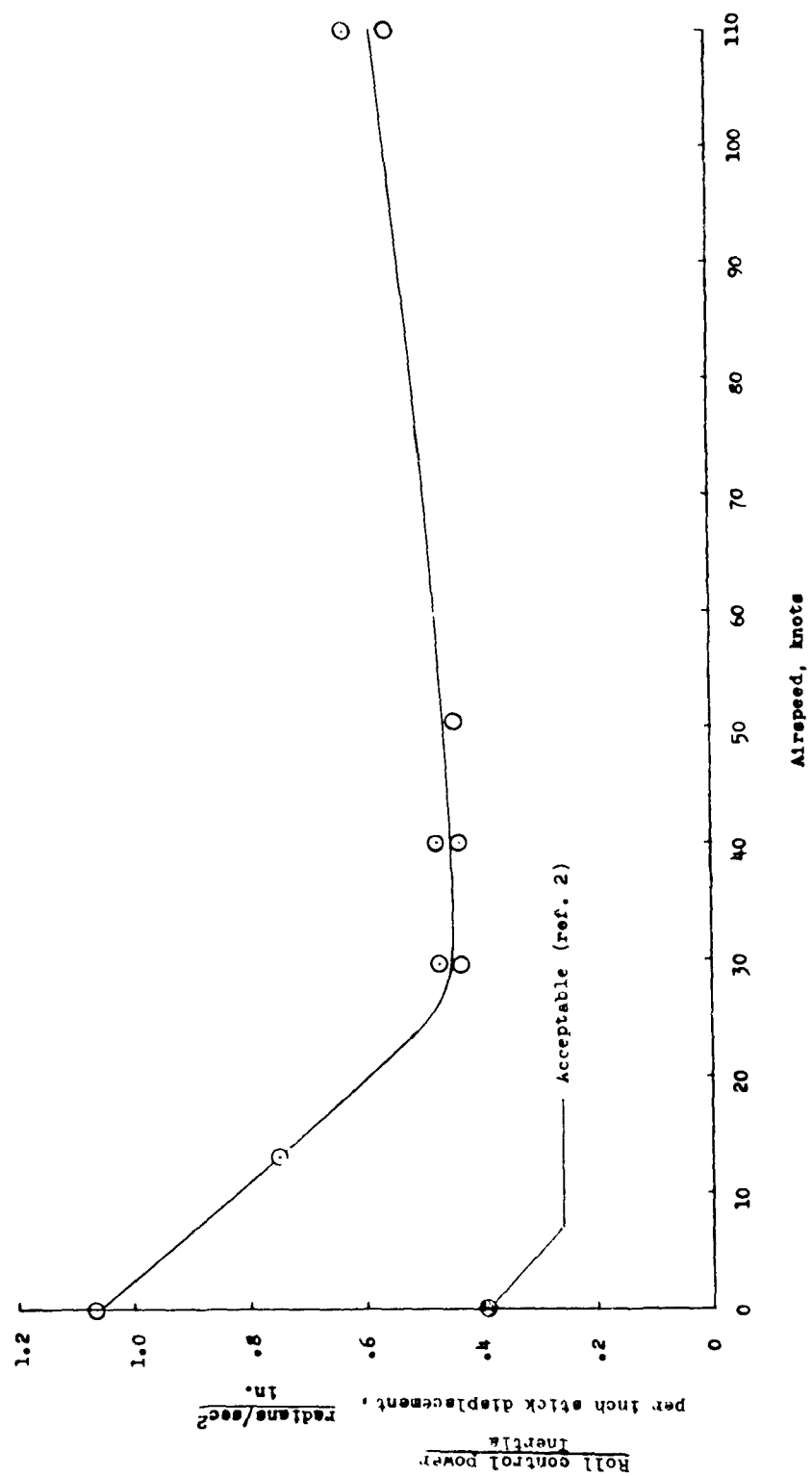


Figure 14.-- Roll control power.

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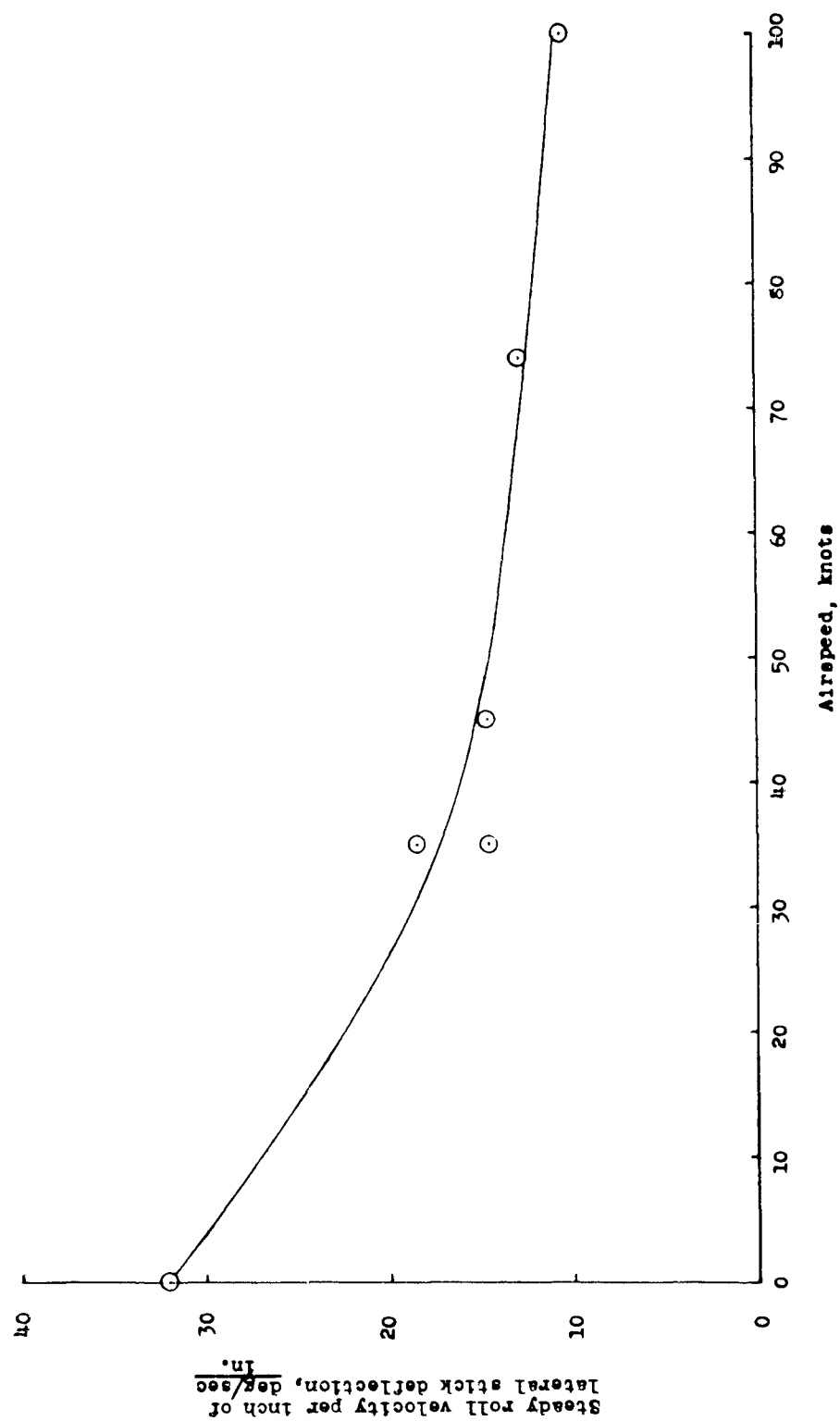


Figure 15.- Roll velocity per inch of lateral stick deflection plotted against airspeed.

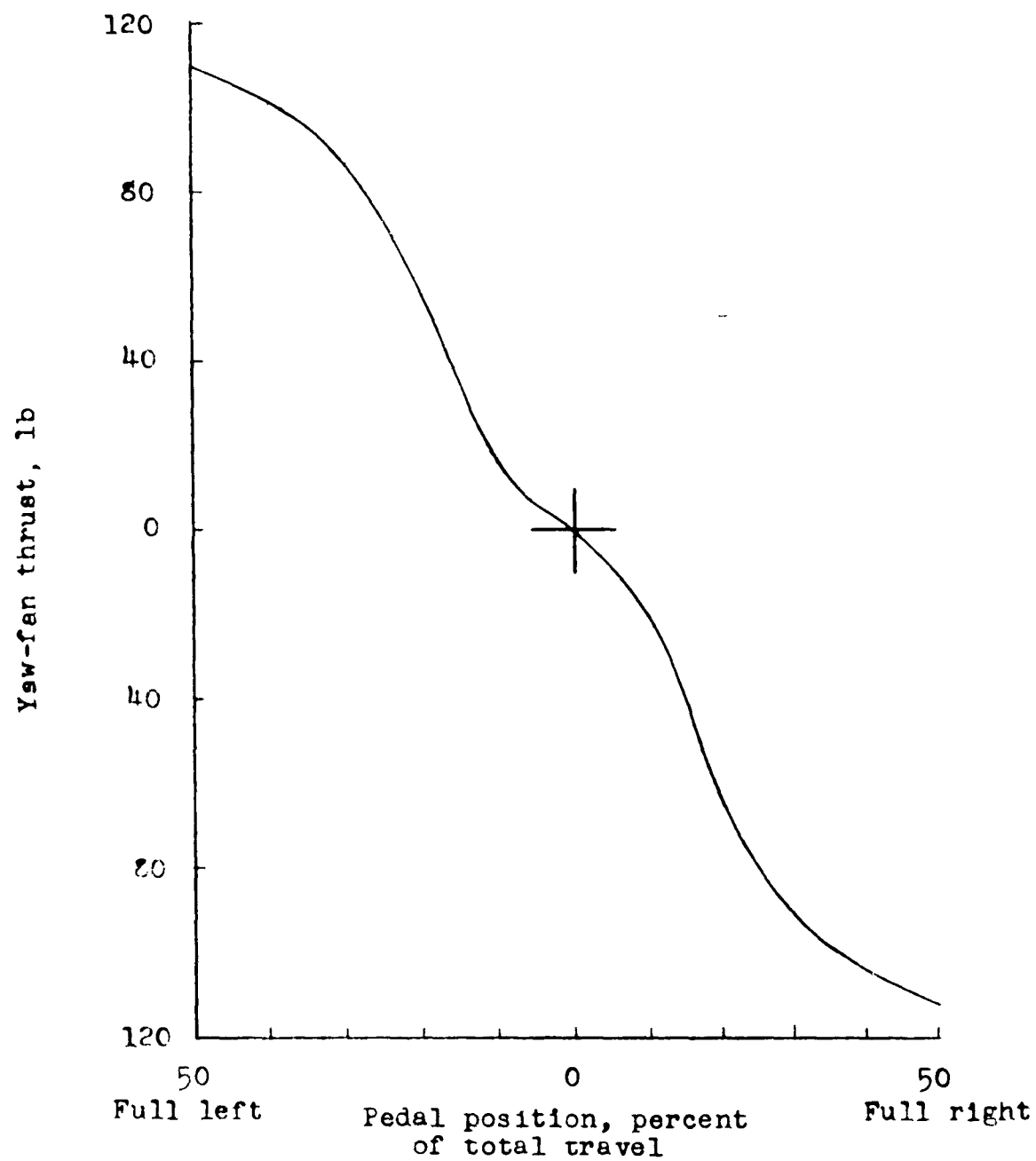


Figure 16.- Variation of yaw-fan static thrust with pedal displacement.

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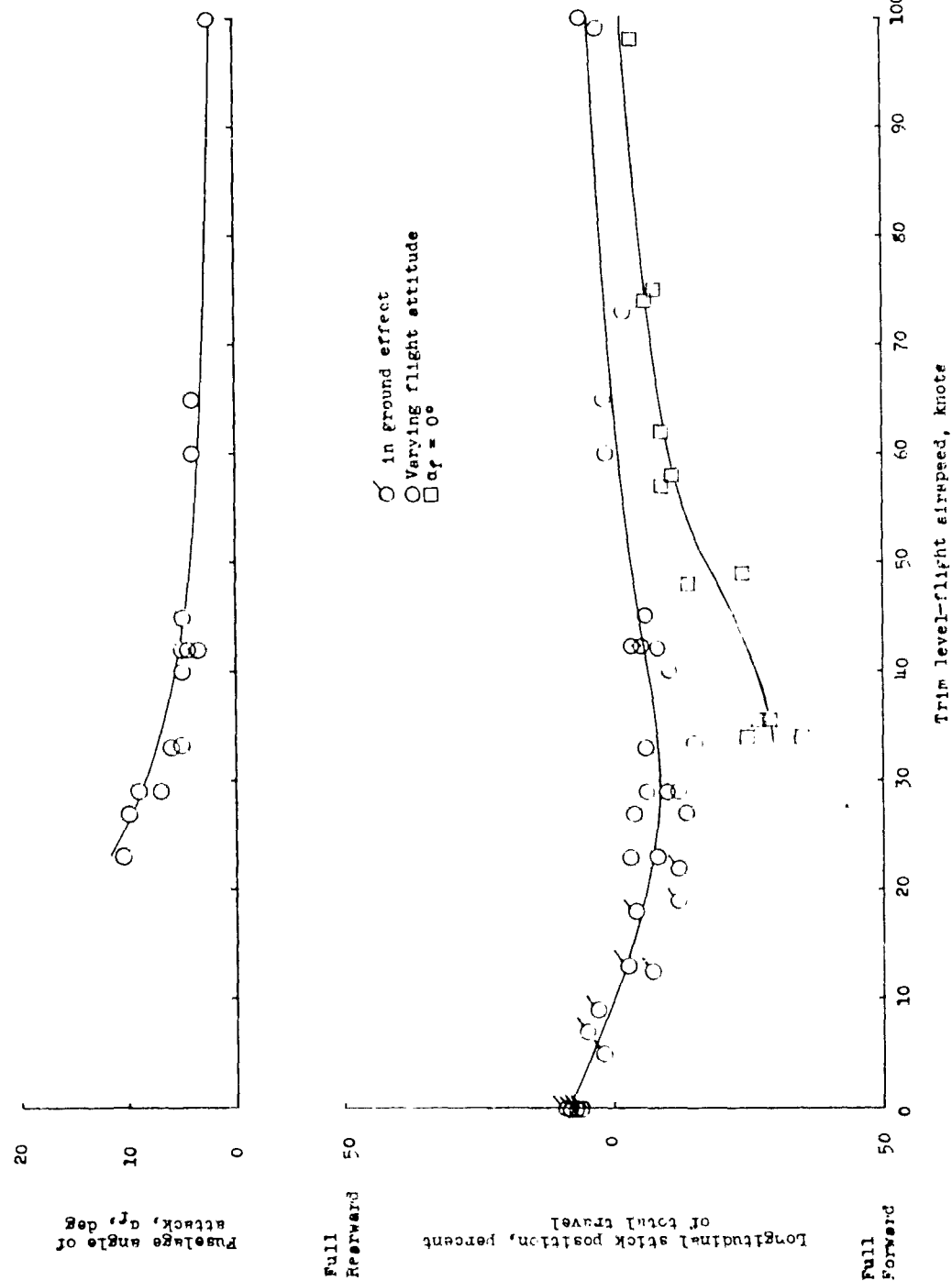


Figure 17.- Variation of longitudinal stick position with airspeed, indicating effects of fuselage attitude as wing angle is varied through 76° .

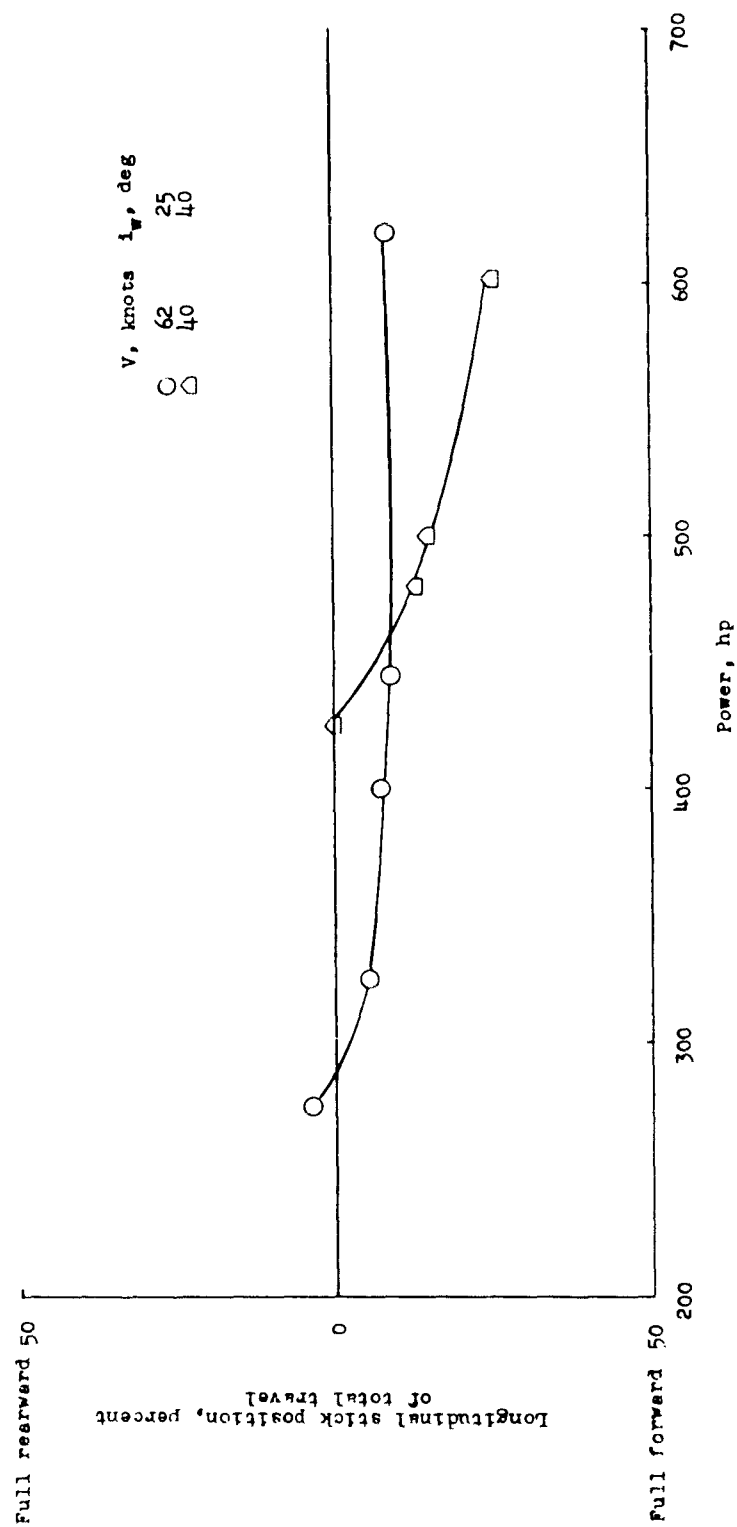
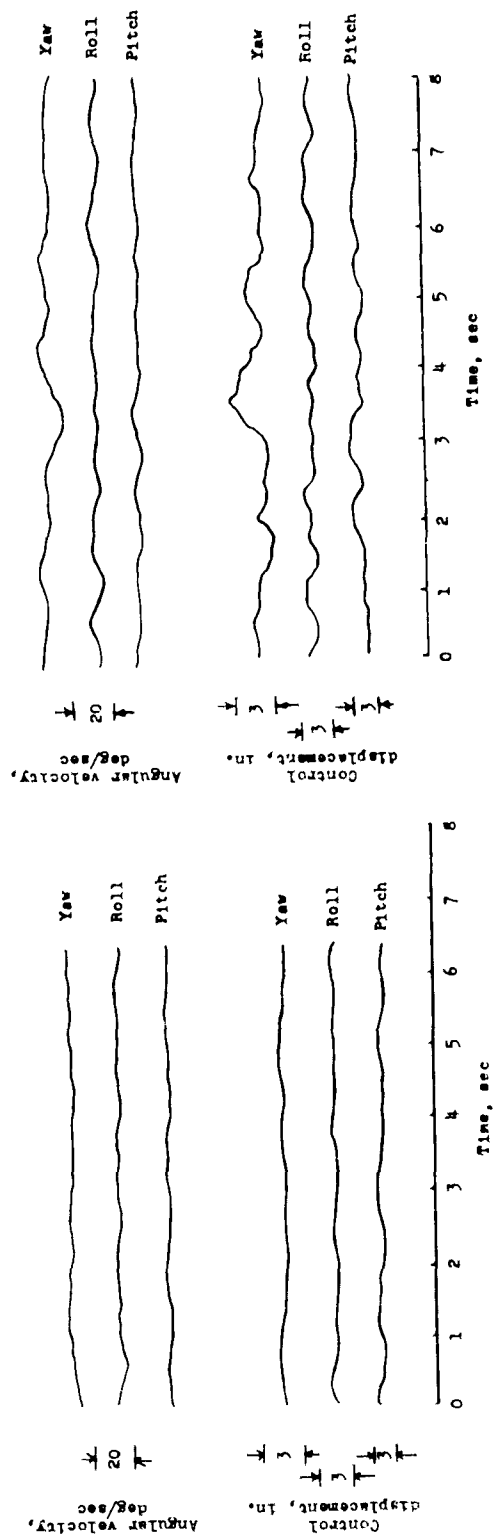


Figure 18.- Variation of longitudinal stick position with power at two different wing angles.



(a) Out of ground effect.

(b) In ground effect.

Figure 19.- Aircraft motion in and out of ground effect without automatic stabilization.
 $i_W = 680$.

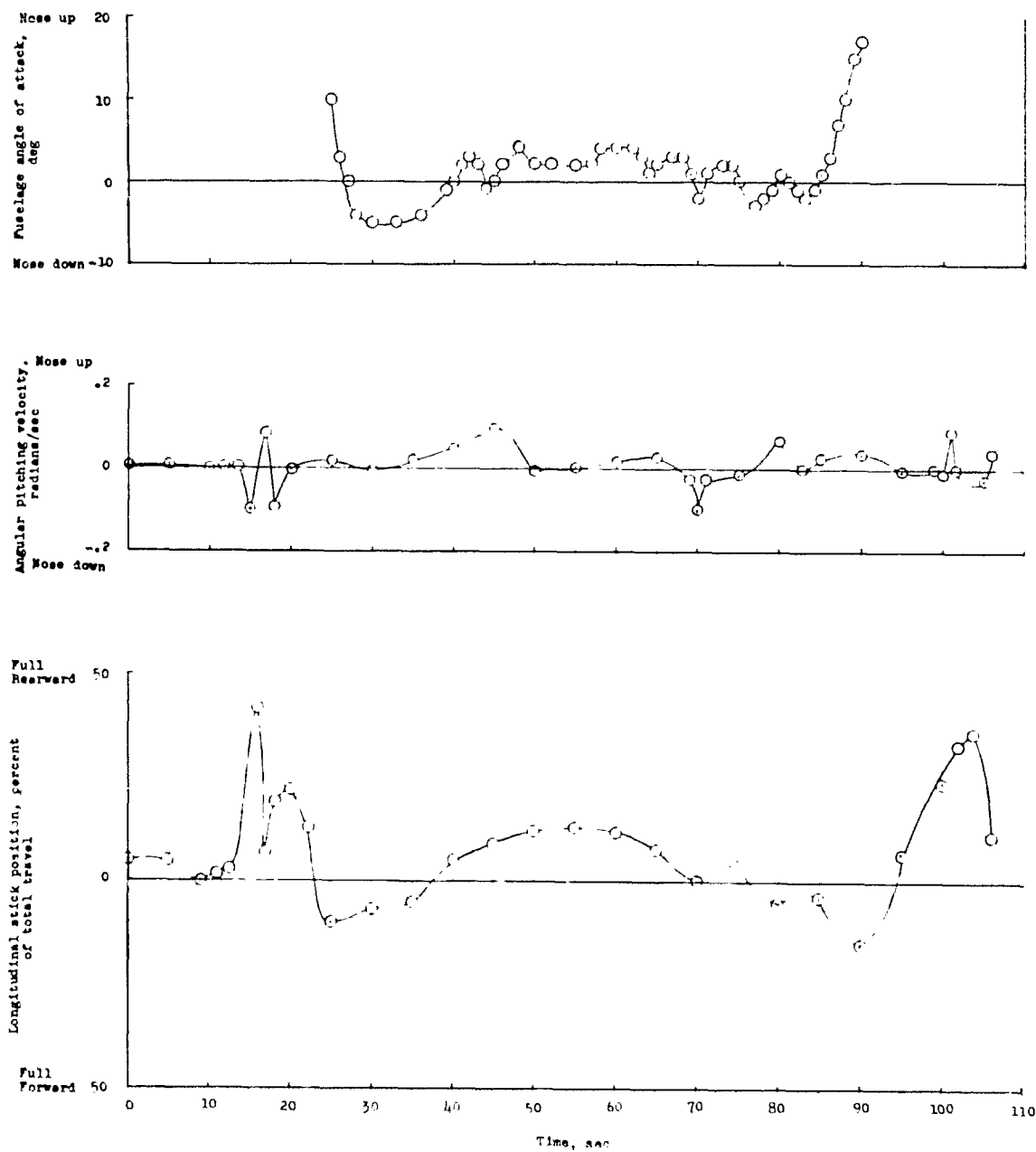


Figure 20.- Time history of typical level-flight conversion performed by the tilt-wing VTOL aircraft.

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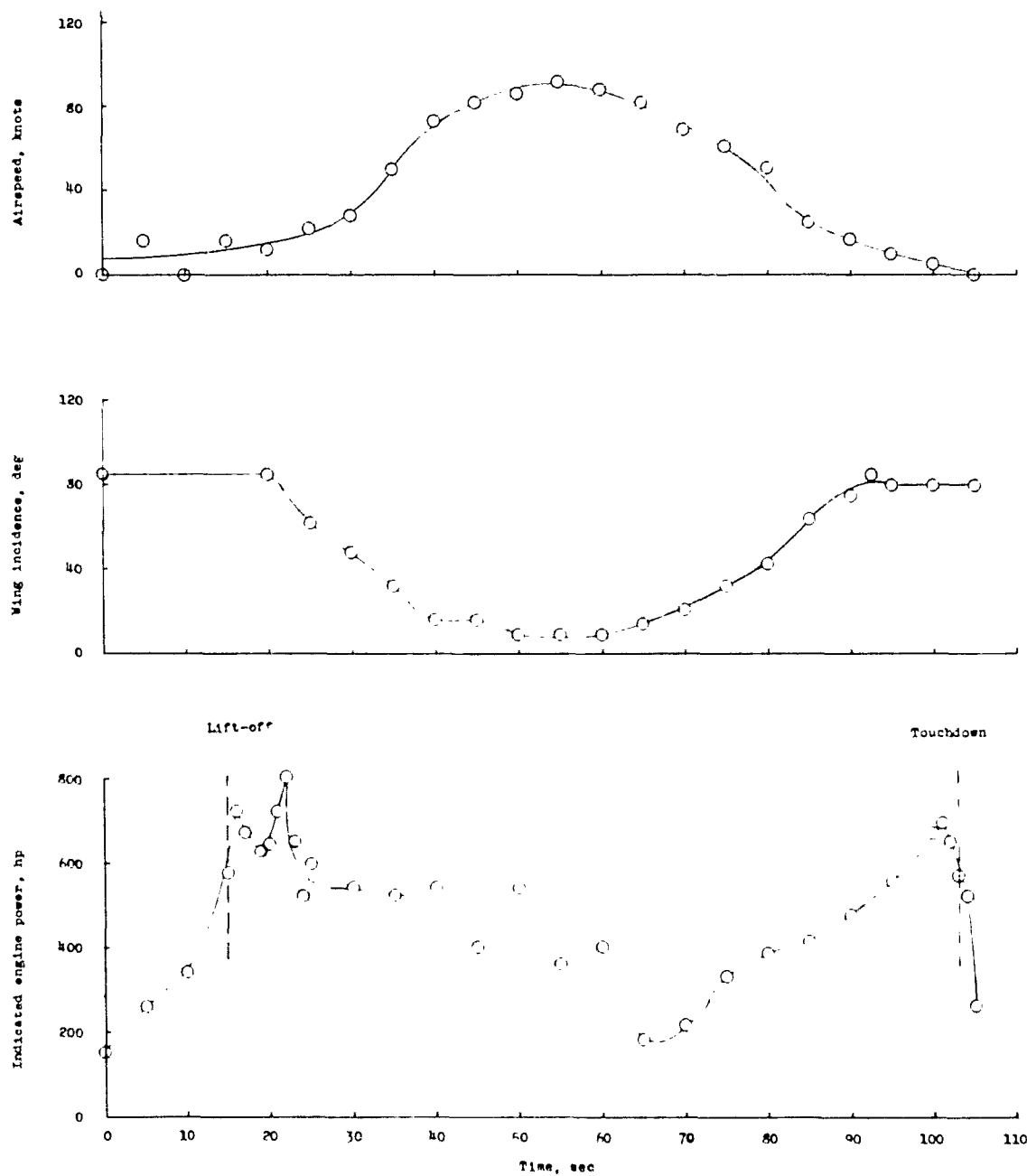
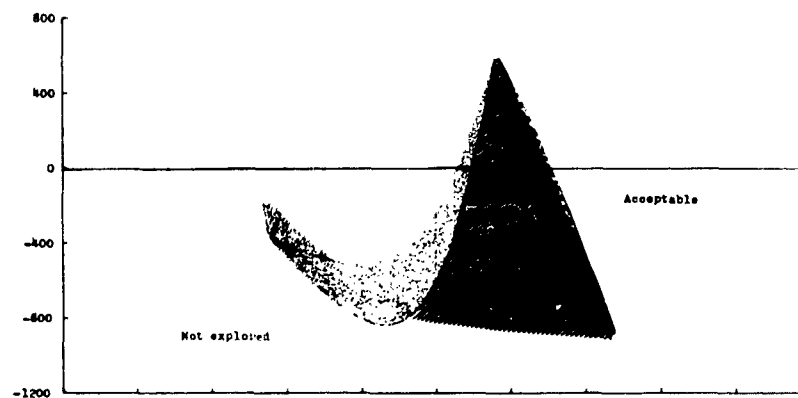
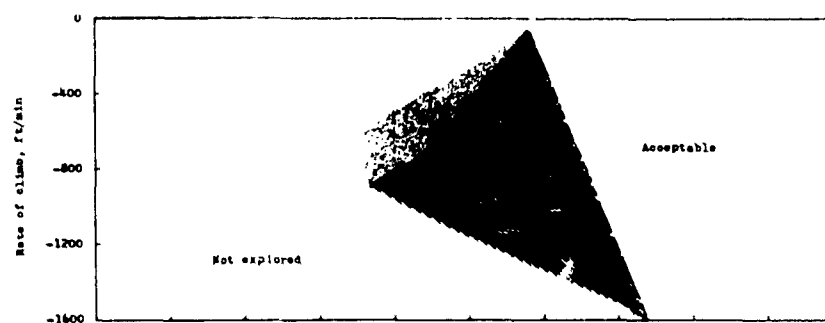


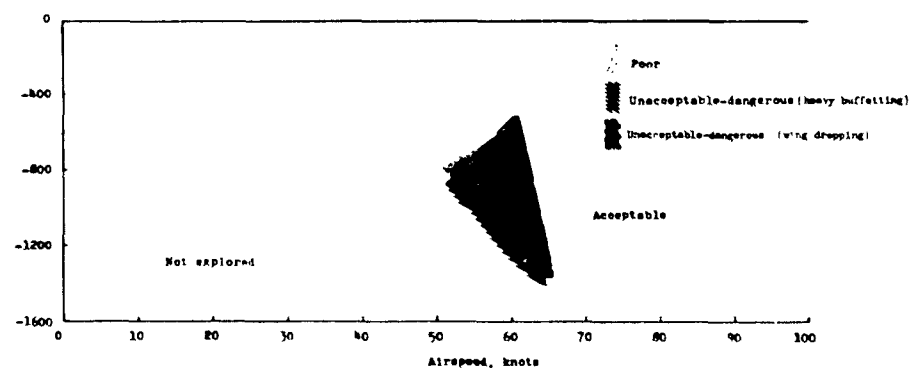
Figure 20.- Concluded.



(a) Aircraft with basic wing.



(b) Aircraft with basic wing with outboard droop.



(c) Aircraft with basic wing with full-span droop.

Figure 21.- Rate-of-descent boundaries.

<p>NASA TN D-989 National Aeronautics and Space Administration. SUMMARY OF FLIGHT-TEST RESULTS OF THE VZ-2 TILT-WING AIRCRAFT. Robert J. Pegg. February 1962. 42p. OTS price, \$1.25. (NASA TECHNICAL NOTE D-989)</p> <p>Flight-test information gained from the research aircraft has shown that design problems exist in such fields as low-speed stability and control, handling qualities, and flow separation during transition. The control power in the near-hovering configuration was considered by the pilots to be inadequate in yaw, marginal in pitch, and excessive in roll. Solutions for some of the design problems are indicated; for example, the addition of a leading-edge droop to the wing in an attempt to delay flow separation resulted in such significantly improved handling qualities in the transition range that an additional descent capability of 1,100 feet per minute was obtained.</p>	<p>I. Pegg, Robert J. II. NASA TN D-989</p> <p>(Initial NASA distribution: 1, Aerodynamics, aircraft; 3, Aircraft; 50, Stability and control; 53, Vehicle performance.)</p>	<p>NASA TN D-989 National Aeronautics and Space Administration. SUMMARY OF FLIGHT-TEST RESULTS OF THE VZ-2 TILT-WING AIRCRAFT. Robert J. Pegg. February 1962. 42p. OTS price, \$1.25. (NASA TECHNICAL NOTE D-989)</p> <p>Flight-test information gained from the research aircraft has shown that design problems exist in such fields as low-speed stability and control, handling qualities, and flow separation during transition. The control power in the near-hovering configuration was considered by the pilots to be inadequate in yaw, marginal in pitch, and excessive in roll. Solutions for some of the design problems are indicated; for example, the addition of a leading-edge droop to the wing in an attempt to delay flow separation resulted in such significantly improved handling qualities in the transition range that an additional descent capability of 1,100 feet per minute was obtained.</p>	<p>I. Pegg, Robert J. II. NASA TN D-989</p> <p>(Initial NASA distribution: 1, Aerodynamics, aircraft; 3, Aircraft; 50, Stability and control; 53, Vehicle performance.)</p>
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